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TEST PROGRAM. EIGHTY-FIVE AND  
100-POUND-THRUST OAMS THRUST CHAMBER  
ASSEMBLIES FINAL REPORT (Rocketdyne) 134 p

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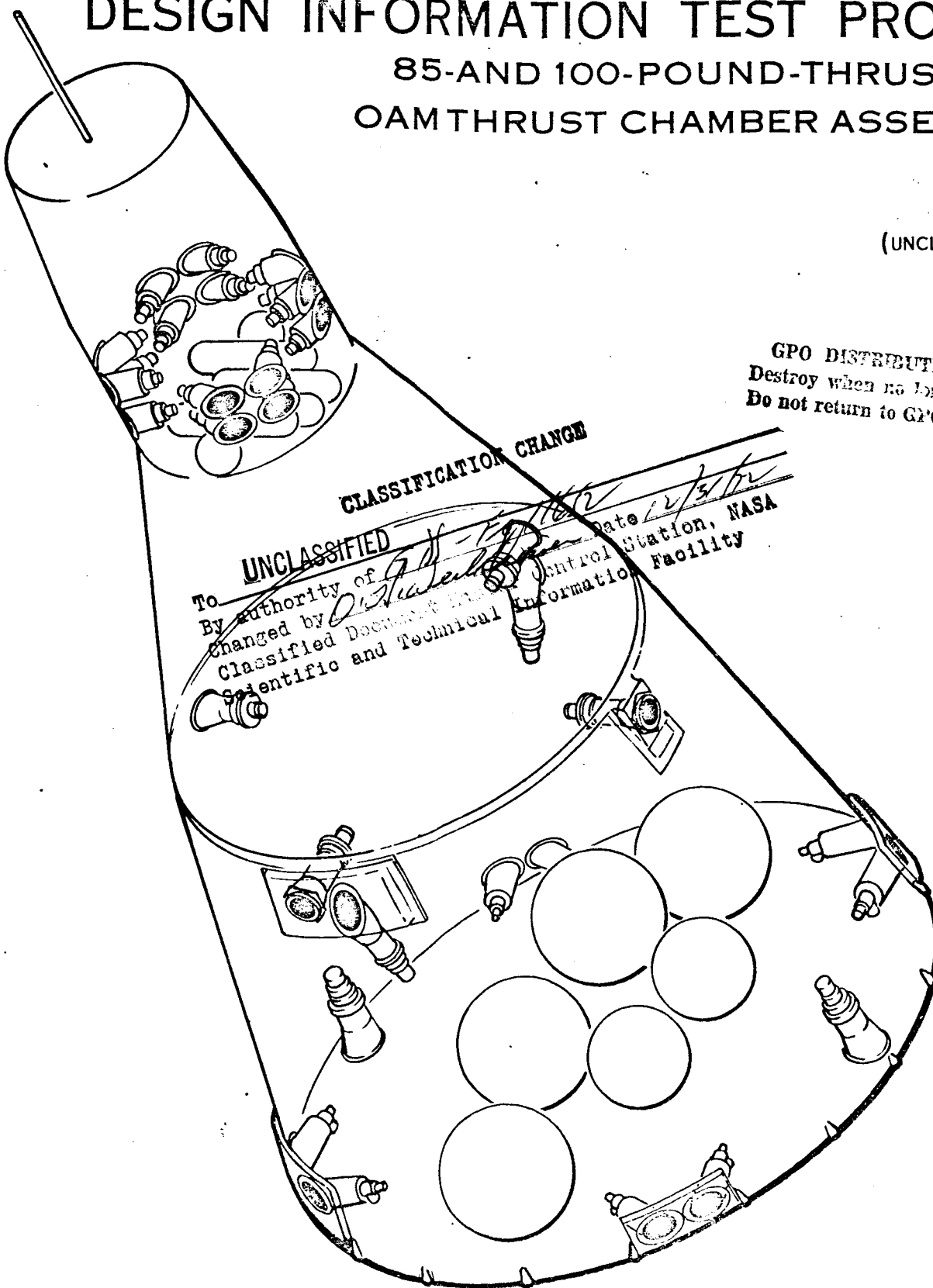
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*P.A.* **GEMINI**  
**DESIGN INFORMATION TEST PROGRAM**  
**85-AND 100-POUND-THRUST OAMS**  
**OAM THRUST CHAMBER ASSEMBLIES**

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(Unclassified Title)

GEMINI DESIGN INFORMATION TEST PROGRAM

FINAL REPORT

85- AND 100-POUND-THRUST OAMS

THRUST CHAMBER ASSEMBLIES

~~Group 4~~  
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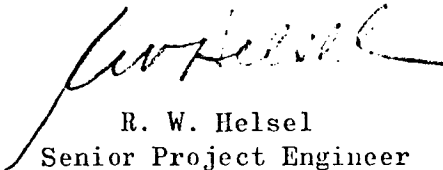
P.O. Y20161R

G.O. 8644

**PREPARED BY**

Spacecraft Engine Division  
Canoga Park, California

**APPROVED BY**

  
R. W. Helsel  
Senior Project Engineer  
Gemini

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#### FOREWORD

This report on the Design Information Testing of Gemini 85- and 100-pound-thrust OAMS Thrust Chamber Assemblies was prepared for the McDonnell Aircraft Corporation in accordance with the requirements of Purchase Order Y20161R.

#### ABSTRACT

The Design Information Test program conducted on Gemini 85- and 100-pound-thrust OAMS Thrust Chamber Assemblies is described. Test data, photographs of test equipment, and test summaries are included.

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## INTRODUCTION

This final report presents the Design Information Testing (DIT) of the Gemini 85- and 100-pound-thrust Orbit and Attitude Maneuvering System (OAMS) Thrust Chamber Assemblies (TCA's). The purpose of DIT was to explore the capability of the thrust chamber to operate under more adverse conditions than normally encountered in flight.

DIT was conducted on production configuration TCA's except for (1) the propellant valve inlet connections, and (2) the use of chamber pressure taps to measure combustion chamber pressure during the tests. The inlet fittings incorporated threaded connectors in place of tube stubs supplied with production engines. The chamber pressure tap was utilized during test firings; however, it is not used in flight. Propellant valves were actuated by a 26-volt d-c power source with no spike-suppression circuits.

This report summarizes results from the individual tests and provides added interpretation in a manner similar to that which was done for the 25-pound-thrust Re-entry Control System (RCS) and 25-pound-thrust OAMS TCA's\*. A brief recapitulation of each test is presented prior to conclusions. Performance and durability aspects are reviewed separately. Also, operating conditions likely to cause catastrophic failures are discussed, as is testing which would contribute additional information on the capability of the 85- and 100-pound-thrust TCA's.

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\*R-15038-1, Final DIT Report, 25-pound RCS and 25-pound OAMS Engines

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### CONCLUSIONS

DIT has been extremely worthwhile in evaluating the capability of the 85- and 100-pound-thrust TCA's. One 85- and one 100-pound-thrust TCA were subjected to a steady-state endurance test to failure, proving that both models will exceed specification requirements. Results of the low supply pressure, low-temperature, and stuck valve tests indicate no conditions that might be catastrophic other than the obvious loss of propellant encountered with a propellant valve stuck open. The off-mixture ratio test indicated capability of operating safely at all but the +50 percent of nominal mixture ratio condition. The TCA cannot withstand operating under this condition very long before exceeding a 600 F skin temperature and eroding severely. The plugged injector test was considered to have been exceptionally severe; however, the TCA successfully completed all but the final steady-state part of the OCM 218 MDC with only a 14-percent loss of specific impulse. The high-temperature tests were run for 75 seconds, which indicates that the propellant valves will operate satisfactorily under adverse temperature conditions. This test reveals a potential problem of injector erosion which should be explored further.

It is therefore believed that the design information tests have been of unquestionable benefit in exploring the capability of the 85- and 100-pound-thrust OAMS thrust chambers.

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## PERFORMANCE

### LOW SUPPLY PRESSURE (APPENDIX H)

The purpose of this test was to determine the effect on thrust chamber performance with reduced propellant pressure at the valve inlets. The test simulates what might occur in flight in the event of a regulator malfunction or approaching depletion of pressurant. A regulator malfunction could cause reduced propellant pressure in either of two ways: (1) if the regulator were stuck partially closed and TCA firing demand exceeded pressurant flow capacity and the astronaut elected not to use the bypass valve, or (2) if the bypass valve were used and other duties occupied the astronaut's attention so that he momentarily did not observe propellant tank pressure decay.

The tests were run at test site ambient temperature and pressure. The mixture ratio was set at 1.6 prior to reducing inlet pressure. Thirty-three steady-state firings were made of 5 seconds duration each at inlet pressures of 75, 50 and 25 percent of design inlet pressure, respectively.

The test revealed that both  $c^*$  and mixture ratio remained essentially constant for the 75 and 50 percent of nominal inlet pressure tests, but dropped approximately 8 percent at the 25-percent inlet pressure. Also, at the 25-percent point, the mixture ratio was reduced from 1.6 to approximately 1.37. Chamber pressure was reduced progressively as inlet pressure was reduced and reached a low of 30 psig at 25-percent inlet pressure. The test is significant in showing that severe losses in  $c^*$  do not occur with inlet pressure reduced by as much as 50 percent. Additional information could have been obtained if the test had been run at altitude.

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## OFF-MIXTURE RATIO (APPENDIX D)

The off-mixture ratio test was conducted to investigate the effect on TCA performance and life of operating at  $\pm 10$  and  $\pm 50$  percent of nominal mixture ratio. Three firings of 3.2 seconds duration each were made at each of the four mixture ratio settings. Mixture ratio for the  $\pm 10$ -percent points was established by setting the desired mixture ratio and controlling the total flow so that it remained the same as it would for a nominal mixture ratio setting. Mixture ratio for the  $\pm 50$ -percent test was established by reducing oxidizer flow to obtain the -50-percent setting and decreasing fuel flow to obtain the +50-percent setting. In these cases, total flow was not held constant. In addition to the calibration test, and orbit component maneuver (OCM) 218 mission duty cycle (MDC) was conducted at the high mixture ratio setting to evaluate durability. This will be discussed subsequently.

Curves of  $c^*$ , specific impulse, and  $C_F$  for each mixture ratio setting are contained in Appendix D.  $C_F$  remained essentially constant over the range of mixture ratios tested;  $c^*$  dropped about 4 percent for a 10-percent reduction in mixture ratio and 13 percent for a 50-percent reduction in mixture ratio. Optimum  $c^*$  appeared to occur at a 2:1 oxidizer-fuel ratio. Specific impulse increased approximately 1.5 percent above nominal at the high mixture ratio setting, and decreased approximately 12.5 percent at the low mixture ratio setting. Theoretical optimum mixture ratio for frozen equilibrium occurs at approximately a 1.8 mixture ratio.

These performance relationships show that large increases in mixture ratio above design nominal have relatively small performance effects. On the other hand, large reductions in mixture ratio below design

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nominal have a pronounced effect on performance. The variation in performance between TCA's is not taken into consideration for the results noted. In all probability, these performance variations will be as large as the variations in performance experienced in the mixture ratio range of  $\pm 10$  percent of nominal.

#### PLUGGED INJECTOR (APPENDIX G)

The plugged injector test was a test performed with four sequential oxidizer holes plugged and four sequential fuel holes plugged (nonimpinging). The two rows of plugged holes were on the same side of the injector separated by one pair of impinging holes (Fig. 32 of Appendix G). This was considered to be the most severe manner which the injector might be plugged. The test was run to determine what performance and physical effects would result from operating in this condition. The test was designed to evaluate a potential malfunction although it is almost out of the realm of possibility to have 25 percent of both oxidizer and fuel passages blocked in flight. The test is considered extremely severe.

The performance of the TCA prior to and after plugging the injector holes is shown in Table 2 of Appendix G. For comparable propellant inlet pressures, mixture ratio increased from 1.61 to 1.77, chamber pressure dropped from 148 to 132 psig, and  $c^*$  dropped from 5221 to 4489 (14 percent) as a result of plugging the injector. The flame pattern was severely disturbed as evidenced by the highly localized erosion of the chamber following the test.

The TCA performance is considered surprisingly good in view of the severity of operating with 25 percent of both fuel and oxidizer passages plugged.

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TCA transient performance was unaffected. The test dispels any doubt that the TCA would continue to operate satisfactorily if only one or two holes were partially plugged; a condition more likely to be experienced in flight.

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## DURABILITY

### CONTINUOUS TEST TO CHAMBER FAILURE (APPENDIX A)

The purpose of this test was to determine the maximum capability of the 85-pound-thrust TCA when operated continuously.

The test was performed at a simulated altitude of approximately 100,000 feet at ambient temperature. Three calibration tests, plus six 1-second tests simulating prelaunch checkout, were made prior to the start of endurance testing.

Tabulated below is a comparison of specification requirements and test results.

Parameter	Specification Requirements	Test Results
Thrust (maximum), pounds	83	82
Specific Impulse (maximum), seconds	280	296 (approximately)
Throat Skin Temperature (after total firing time of 147 seconds), F	500	180
Throat Skin Temperature (after total firing time of 270 seconds*), F	700	300

\*Includes calibration plus Cape cycle time

The test was extended beyond the minimum specification requirement of 270 seconds to 358.6 seconds of steady-state operation or 373.6 seconds total

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time including calibration and cycle time. Throat erosion started after 40 seconds of operation and gradually increased as the test progressed, as evidenced by the curve of chamber pressure vs time. The test was terminated because of burnthrough at the entrance to the throat.

The test shows that the 85-pound-thrust TCA exceeds specification requirements. Further, it shows that the point of catastrophic failure is well beyond the specification requirement of 270 seconds. The sectioned chamber is shown in Fig. 6 of Appendix A. Although severely eroded, the erosion pattern is uniform, indicating a good injector spray pattern.

#### CONTINUOUS TEST TO CHAMBER FAILURE (APPENDIX B)

Essentially the same test as described above was conducted using a 100-pound-thrust TCA.

The test was conducted at an altitude of approximately 100,000 feet and at ambient temperature. The chamber received three calibration tests, plus six 1-second tests simulating prelaunch checkout prior to the endurance testing.

The following is a tabulation of test results compared to specification requirements.

Parameter	Specification Requirements	Test Results
Thrust (maximum), pounds	99.3	102
Specific Impulse (maximum), seconds	280	293 (approximately)

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Parameter	Specification Requirements	Test Results
Throat Skin Temperature (after total firing time of 220 seconds*), F	650	180
Throat Skin Temperature (after total firing time of 300 seconds*), F	700	200

\*Includes calibration plus Cape cycle time

With exception of thrust being slightly in excess of the specification maximum, the performance of the TCA was within specification limits for the entire test. Failure started at 425 seconds, but the test was allowed to continue until burnthrough occurred. The TCA steady-state operating time was 443.8 seconds with a total firing time (including calibration and cycle time) of 459.4 seconds.

A posttest examination of the thrust chamber revealed two burnthrough areas at the injector end of the chamber. Also, there was uneven erosion in the combustion chamber and throat area. Part of the splashplate was burned away which affected the injector spray pattern and may have caused the uneven erosion of the combustion chamber. A detailed evaluation of the condition and probable cause of failure is contained in Appendix B. Despite the posttest condition of the TCA, there was no catastrophic failure (as defined) within the specification requirement.

Steady-state operation is the most severe type of operation from the standpoint of erosion. However, if flight events require steady-state firing up to 270 seconds, the results of this test indicate that the 100-pound-thrust TCA can accommodate such a requirement with a margin of safety.

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## EXTREME LOW-TEMPERATURE TEST (APPENDIX C)

The low-temperature test was conducted by precooling the TCA to -100 F prior to each firing. Propellants were maintained above +15 F during the tests. Five steady-state firings were made with inspection of the TCA following each test and recooling prior to the start of the following test.

The purpose was to simulate temperatures which might be encountered in flight to determine if start transients would be affected and if thermal shock would cause damage to materials within the TCA. There was no requirement to investigate freezing of propellants within the injector passages; therefore, this facet was not tested.

The chamber tested had two longitudinal cracks in the throat insert and two cracks in the second liner segment prior to the start of cold testing.

Oscillograph records indicated no lag in start time as a result of the -100 F temperature of the TCA and the +15 F temperature of the propellants. Posttest inspection revealed no additional damage to the liner segment which was previously cracked. The throat, however, sustained two additional cracks; one longitudinal and one circumferential just forward of the minimum diameter. Erosion of the throat was encountered in the area between the two cracks. X-ray examination revealed no damage to ablative materials.

It is not clear whether the additional throat cracks sustained during the tests were related to the -100 F initial chamber temperature. The fact that the throat was already cracked at the start of the test might be indicative of a throat with inferior physical properties.

Despite the cracks, the throat insert remained intact and the five 15-second-duration tests were completed successfully. The ablative materials were in excellent condition at the conclusion of the test as indicated by X-ray examination.

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#### OFF-MIXTURE RATIO TEST (APPENDIX D)

The effect on performance of shifting mixture ratio  $\pm 10$  percent and  $\pm 50$  percent from nominal has been previously reported. Discussed herein are the results of conducting an OCM 218 MDC with mixture ratio set 50 percent above nominal (2.4 oxidizer/fuel ratio).

The specification maximum skin temperature of 600 F was exceeded during the off period between the fifth and sixth pulses of the MDC which occurred after 197 seconds of operation. The temperature increased gradually reaching a value of 750 F at cutoff at the completion of the final 95 seconds of steady-state operation following the MDC. Thrust and chamber pressure started to decay during the last 5 seconds of the first 15-second firing. Although the JTA liner and throat were consumed completely and the combustion zone was heavily glassed, the erosion was uniform and there was no burnthrough. The chamber was charred completely, but there were no severe delaminations.

The test indicates that the chamber operating at a 2.4 mixture ratio will not stay within specification requirements throughout an OCM 218 MDC. The temperature experienced at this mixture ratio is in excess of that which the JTA liners and silicon carbide throats will withstand. Probably, the decrease in chamber pressure helped to prevent delamination of ablative materials and burnthrough.

#### STUCK VALVE TEST (APPENDIX E)

The purpose of the stuck valve test was to determine the hazard involved in operating a TCA with a fuel valve stuck open and the oxidizer valve

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operating, and vice versa. The test was conducted on a 100-pound-thrust TCA at local ambient temperature and pressure. The TCA was mounted with the nozzle inclined upward at an angle of 17 degrees to make the propellant puddle in the injector splashplate. One of the combustion zone liner segments was cracked prior to the test.

During the stuck fuel valve test, 11.8 pounds of MMH were consumed in 79.25 seconds of 16 pulses. During the stuck oxidizer valve test, 13.44 pounds of nitrogen tetroxide were consumed in 58.37 seconds of 12 pulses. In both cases, the TCA was fired 2.5 seconds and allowed to remain off for 3.5 seconds for each pulse.

During the off period of the pulse cycle, propellant flow from the open valve increased because of the lack of combustion chamber pressure. An increase of approximately 40 percent was encountered on the fuel side and 22 percent on the oxidizer side. TCA response time was consistent and there were no abnormal oscillations of chamber pressure or temperature during the test. Chamber  $c^*$  efficiency was also not affected.

A posttest examination of the TCA revealed no change in the condition of the liner segment which was cracked prior to the test. X-ray examination indicated several delaminations of the ablative body.

The test showed that the only significant effect of operating with a propellant valve stuck open is the relatively large increase in propellant consumption when the chamber is not operating. Possibly, an extension of the off-time of the duty cycle could produce significantly different results. As an example, if larger quantities of oxidizer were allowed to enter the hot TCA, erosion might increase. Additional testing would be necessary to determine if this would happen.

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## HIGH-TEMPERATURE TEST (APPENDIX F)

A test was conducted with a 100-pound-thrust TCA preconditioned by heating to 200 F. Propellants were heated to 165 F prior to the test and maintained at this temperature throughout the test series. The test was conducted at an altitude slightly in excess of 100,000 feet. The purpose of the test was to determine the effect on TCA durability of preheating the TCA and operating with propellants at 165 F. The test was conducted in the following manner: (1) the chamber was fired for 30 seconds continuously followed by a 5-minute off-period to permit the propellant valve seat temperature to reach a maximum value; (2) the first cycle was followed by two similar cycles, except firing time was 20 seconds in each case; (3) at the third cycle, the TCA was pulsed 10 times, 0.5 seconds on, followed by 10 seconds off.

The thrust chamber used for the tests had previously completed 65.6 seconds of stuck valve design information testing. The throat had sustained two radial cracks prior to the start of the high-temperature test. One JTA liner had a longitudinal crack adjacent to the high-response chamber pressure pickup hole.

The purpose of the test was to learn the effect of high environmental temperature and propellant temperature on the life of the TCA and operation of propellant valves.

The test was completed satisfactorily and operation of the TCA was normal. However, posttest inspection disclosed erosion of the injector and six of the seven liner segments were missing. Heavy deposits of liner material were noted in the region immediately downstream of the throat. The throat insert had additional cracks but the minimum diameter was not eroded.

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There were also numerous delaminations of ablative material; however, the chamber was not burned through. Total time on the TCA including previous testing was 147 seconds.

From the posttest condition of the chamber, it is apparent that high-temperature operation is more severe than testing with room temperature propellants and environment: how much more severe cannot be determined by a test of this nature. The test does show, however, that the chamber can withstand limited exposure to such conditions. In addition, it shows no abnormal performance effects resulting from operating with the TCA preheated to 200 F and with propellants at 165 F.

#### PLUGGED INJECTOR TEST (APPENDIX G)

Performance of the TCA tested with the plugged injector has been previously discussed. The following discussion will concern the effect on TCA life.

To provide a brief recapitulation of the test, four oxidizer and four fuel holes (nonimpinging) were plugged to simulate a condition which could be created by contamination of the propellant system. The holes selected were all on one side of the TCA, which is considered the most severe condition.

The chamber was tested to an OCM 218 MDC. Following the MDC, a steady-state firing of 95.5 seconds was planned; however, failure of the TCA occurred by burnthrough at the entrance of the throat insert station after 47.16 seconds. As expected, the erosion pattern was all on one side of the chamber. One half of the throat insert was washed away as

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well as the edge of the last few liners. Minor delaminations were present in other parts of the ablative material. Deposits of JTA liner material were found at the throat exit. The skin temperature suddenly rose to 600 F during the steady-state firing at the end of the MDC. Burnthrough was also sudden and covered a large area. Total burn time was 370.86 seconds.

The test revealed that the complete specification duration of 400 seconds cannot be accomplished with 25 percent of the injector holes plugged. Although the test does not prove this, it might be suspected that minor stoppage of injector holes would hardly be noticed.

#### LOW SUPPLY PRESSURE (APPENDIX H)

The primary purpose of the low supply pressure test was to investigate the effect of reduced propellant inlet pressure on performance. This was previously discussed under the performance section. The secondary purpose was to determine the effect on durability.

Twenty-three tests of 5-seconds duration each were run at 75, 50, and 25 percent of nominal propellant supply pressure. The tests were run in descending order of pressure. At the termination of the tests, a posttest inspection revealed that three of the JTA liner segments from the injector end of the chamber were missing. These liners were loose at the start of the tests, which may have been influential in causing them to deteriorate.

The tests were not of sufficient duration to obtain conclusive information on life of the chamber; however, because mixture ratio decreased at

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25-percent supply pressure and did not change appreciably at 50- and 75-percent supply pressure, it may be reasoned that reductions in inlet pressure would not affect TCA life appreciably.

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### CATASTROPHIC FAILURES

Of all items tested, only a few are considered severe enough to be likely to cause premature catastrophic failure.

Premature catastrophic failure of the 100-pound-thrust TCA occurred while performing an OCM 218 MDC at a 2.4 mixture ratio. Shell skin temperature exceeded 600 F between the fifth and sixth pulse after 197 seconds of operation. In addition, decay in both chamber pressure and thrust began after approximately 10 seconds of the first 15 seconds of the MDC. The TCA was allowed to complete the MDC because burnthrough was not experienced. Skin temperature at completion was 750 F. Posttest inspection revealed severe erosion. It is therefore concluded that the 100-pound TCA will not withstand operation at high mixture ratio for an extended period of time without early performance loss caused by throat erosion. The limiting mixture ratio was not defined by the test because there was no testing accomplished between +10 percent and +50 percent of nominal.

Operation of the 100-pound-thrust TCA with 25 percent of the injector holes plugged also produced premature catastrophic failure. Burnthrough in the combustion zone occurred after 47.16 seconds of the last 95.5 seconds of steady-state operation at completion of the OCM 218 MDC. Although skin temperature did not exceed 600 F until the period of steady-state operation was started, the TCA was severely eroded in a highly localized area and failure occurred by a sudden breakthrough covering a rather large area.

A MDC was not conducted as part of the high-temperature test, which makes it difficult to predict if premature catastrophic failure would

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be encountered under these conditions. The chamber was operated for 75 seconds with long heat-soakback periods between firings. The long heat-soakback periods permit the injector to heat which, combined with high-temperature propellants, probably causes burning within the splashplate area as evidenced by splashplate erosion. If allowed to continue, the splashplate would erode to such an extent that the flame pattern in the combustion zone would expand into a cone resulting in direct impingement on the combustion zone walls. This would cause premature failure.

These are the only conditions tested during DIT which either produced or seemed likely to produce premature catastrophic failures.

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#### ADDITIONAL TESTING

Unless such testing has already been accomplished, it would be of interest to define limiting conditions for freezing of propellants within the injector, or provide for maintaining the injector above +15 F for all conditions of flight. It is understood that propellants and propellant valves will be kept above +15 F; however, during long missions it is conceivable that the injector may become cold enough to cause propellants to freeze, particularly if propellant leakage is encountered.

It would be desirable to perform additional evaluation of the effects of stuck propellant valves by exploring longer periods of time between pulses to accumulate larger quantities of propellant or propellant vapors within the TCA. The leaking propellant might possibly flash boil and reach a limiting concentration; however, it would be of interest to determine if such a condition occurs.

It would also be informative to conduct a test with less than 25 percent of the injector holes plugged to determine if a MDC can be performed safely.

Because off-mixture ratio endurance was conducted at only one mixture ratio (2.4), it would be desirable to perform endurance testing at a lower mixture ratio to define an upper limit which may be tolerated with safety.

Erosion of the injector splashplate was experienced during two of the design information tests. One case was encountered with the 100-pound TCA during the continuous test to failure; the second case during the

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high-temperature test. The first case was attributed to leakage of hot gas around the injector, which increased the injector temperature causing propellants to burn within the splashplate area. The second case combined with the high-temperature propellants was attributed to heat soak-back, which caused the injector to overheat. If such a condition occurs after the required life of the chamber has been expended, it is of little concern; however, if a combination of circumstances can cause such a failure early in the life of the TCA, it is of great concern.

It is believed worthwhile to perform additional high-temperature tests with realistic mission conditions taken into consideration to determine if injector overheating and resultant erosion constitute a flight hazard.

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APPENDIX A

CONTINUOUS TEST OF 85-POUND-THRUST  
TCA TO CHAMBER FAILURE

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TITLE: CONTINUOUS TEST TO CHAMBER FAILURE DESIGN INFORMATION TEST

### INTRODUCTION

A continuous firing to catastrophic chamber failure design information test was conducted on 22 May 1964, with 85 lb DIT Unit No. 1 flightweight thrust chamber assembly. The test was conducted at CTL-III, Cell 39D, at a pressure altitude of 100,000 feet or higher and at local ambient temperature. Catastrophic failure is defined as (1) loss of capability to stop propellant flow, (2) any flow or leakage of propellants or combustion products from the TCA at locations other than the TCA exit, and (3) thrust chamber exterior wall or mounting attachments temperature in excess of 600°F.

### SUMMARY

The 85 lb DIT Unit No. 1 flightweight thrust chamber assembly was subjected to a continuous steady state firing to catastrophic failure following three calibration tests and six cape cycle tests. The test was terminated after 358.6 seconds of continuous firing because of a burnthrough of the metal shell of the TCA. In addition to the burnthrough, the throat insert and adjacent ablative in the combustion chamber were severely eroded, resulting in a decay of chamber pressure and thrust increase. This erosion commenced after 40 seconds of continuous firing and continued to increase throughout the test. Temperature of the TCA metal shell in the throat area exceeded 600°F after 347 seconds of continuous firing. Total accumulated burn time was 373.6 seconds, which included 15.6 seconds of calibration and cape cycle firing.

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## DISCUSSION

### Chamber Configuration

The thrust chamber was a standard S/C II production configuration TCA, P/N 207510-41, S/N 4054517, shown on Figure 1. The TCA was manufactured for DIT and did not have any discrepancies that would affect its operation. There were no cracks in the liners or throat insert. The TCA was instrumentated with thermocouples as shown in Figure 2.

### Test Description

The thrust chamber was first calibrated to nominal thrust, mixture ratio, and inlet pressures with a total of three tests for 9.6 seconds of firing time. Following the calibration tests, a series of six one-second duration cape cycle tests were conducted at ambient conditions for 6.0 seconds firing time. The thrust chamber was instrumented with thermocouples as shown in Figure 2 in order to monitor skin and ablative temperatures during the steady state test. The continuous firing steady state test was conducted at a simulated pressure altitude of 120,000 feet.

### Test Results

Calibration Tests - The performance values obtained during calibration testing were: Injector end  $C^*$  of 5330 ft/sec (91.5% theoretical shift-ing) at a mixture ratio of 1.59 and a chamber pressure of 125.0 psia obtained with valve inlet pressures of 285 psia for oxidizer and 287 psia for fuel.  $I_{sp}$  was 16 seconds above nominal.

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Cape Cycle - No data was obtained from the cape cycle tests because the one-second firing duration was not sufficient time for instrumentation to stabilize.

Steady-State Continuous Firing Test - The performance parameters at the start of the test were similar to the calibration tests. As the test progressed, the chamber pressure gradually decayed and the thrust gradually increased (Figure 3). At guaranteed life (270 seconds) thrust was 87 lbs (7 lbs above calibration), chamber pressure was 52 psia, and  $I_{sp}$  was approximately 13 seconds above nominal. After 324 seconds a sharp decrease in thrust and chamber pressure occurred, indicating the start of burnthrough of the ablative. The thrust and chamber pressure decrease continued at a gradual rate until cut-off, indicating a steady increase in the burned out area. The test was terminated after 358.6 seconds of firing time making a total accumulated firing time of 373.6 seconds, including calibration and cape cycle firing.

Temperature increased during the test as shown in Figure 4. At guaranteed life (270 seconds) the temperature of the TCA metal shell was 315°F. As firing continued, the temperature exceeded 1000°F in the area of the localized burnthrough, but this high temperature was not recorded by the thermocouples, which were some distance from the burnthrough and therefore did not immediately reflect the high temperature due to the low thermal conductivity of the stainless steel shell.

Post-test inspection of the thrust chamber revealed that there was a burned through area located at the upstream edge of the throat insert (Figure 5). The throat insert was completely eroded away as was the combustion chamber sleeve and four liners adjacent to the throat insert.

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Examination of the sectioned TCA (Figure 6) shows that there was very little uncharred ablative material remaining in the TCA body, and the 0° wrapped ablative nozzle was delaminated in several places with some loss of the glass fabric. The combustion chamber was severely eroded with no trace of throat insert remaining. The erosion started half-way down the combustion zone removing the four liners adjacent to the throat insert as well as the 0° wrapped sleeve and 90° orientated ablative combustion chamber body. The silica reinforcing material in the ablative body and sleeve was melted and deposited in the nozzle of the TCA. The burn-through was caused by hot gas leakage through a delamination in the heavily eroded portion of chamber body.

#### ANALYSIS

This thrust chamber failed in the expected manner for steady state continuous firing of S/C 2 through 5 TCA's. The injector used with these TCA's has a high heat input to the walls of the combustion chamber and throat insert. Continuous application of heat to the throat insert exceeded the operating temperature of the silicon carbide after approximately 40 seconds of operation and gradually washed the insert away. This combination of continuous high temperature and erosion by the combustion gases also resulted in the deterioration and removal of the JTA liners and combustion chamber sleeve and ablative material as shown in Figure 6. After 324 seconds the chamber ablative material was worn so thin that a delamination occurred in the 90° orientated plies and combustion gases escaped through the delamination burning through the metal shell of the TCA.

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CONCLUSIONS

Guaranteed life of the 85 lb S/C 2 through 5 TCA is 270 seconds, including 131 seconds of operation within specification requirements and additional 123 sec. operation with no catastrophic failure. After firing the 85 lb S/C 2 through 5 configuration TCA for 131 seconds the thrust was 82 lbs, which was within specification limits. The TCA also fired to guaranteed life without catastrophic failure and continued for an additional 90 seconds before a burnthrough occurred. Therefore, it can be concluded that this configuration TCA can be expected to perform satisfactorily for continuous firing to guaranteed life.

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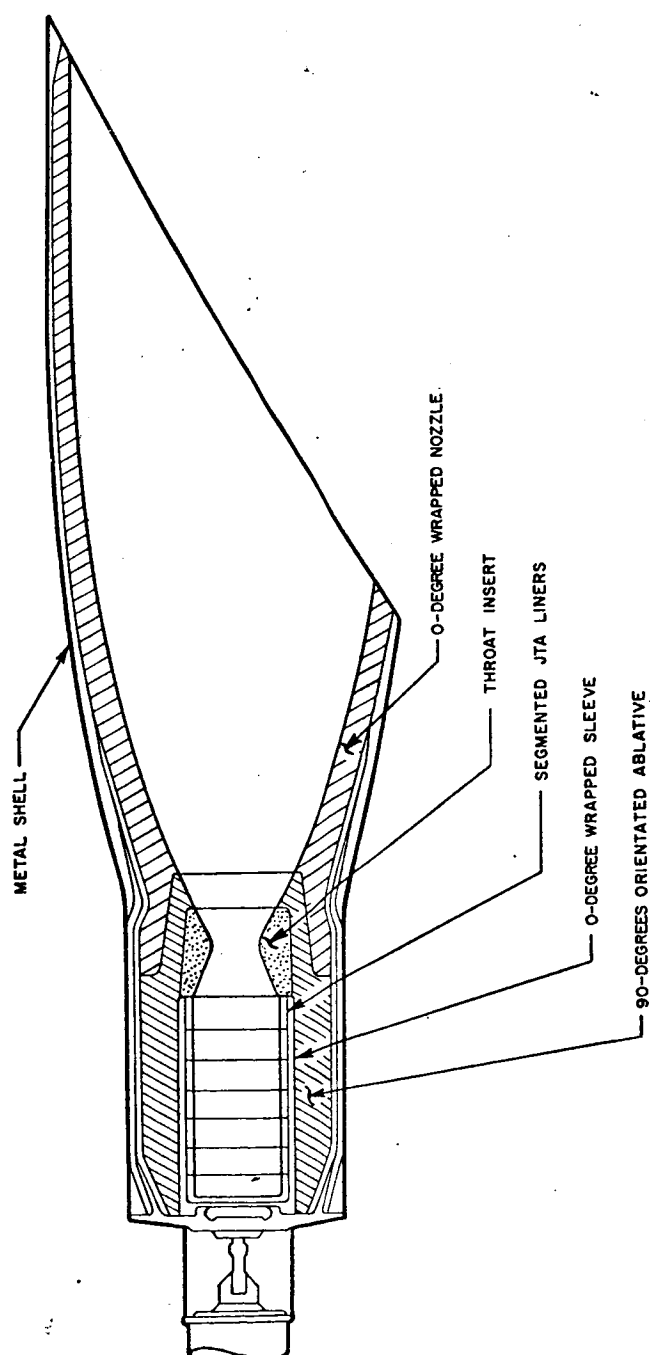


Figure 1. Gemini 85-Pound-Thrust TCA, Cross-Sectional Diagram



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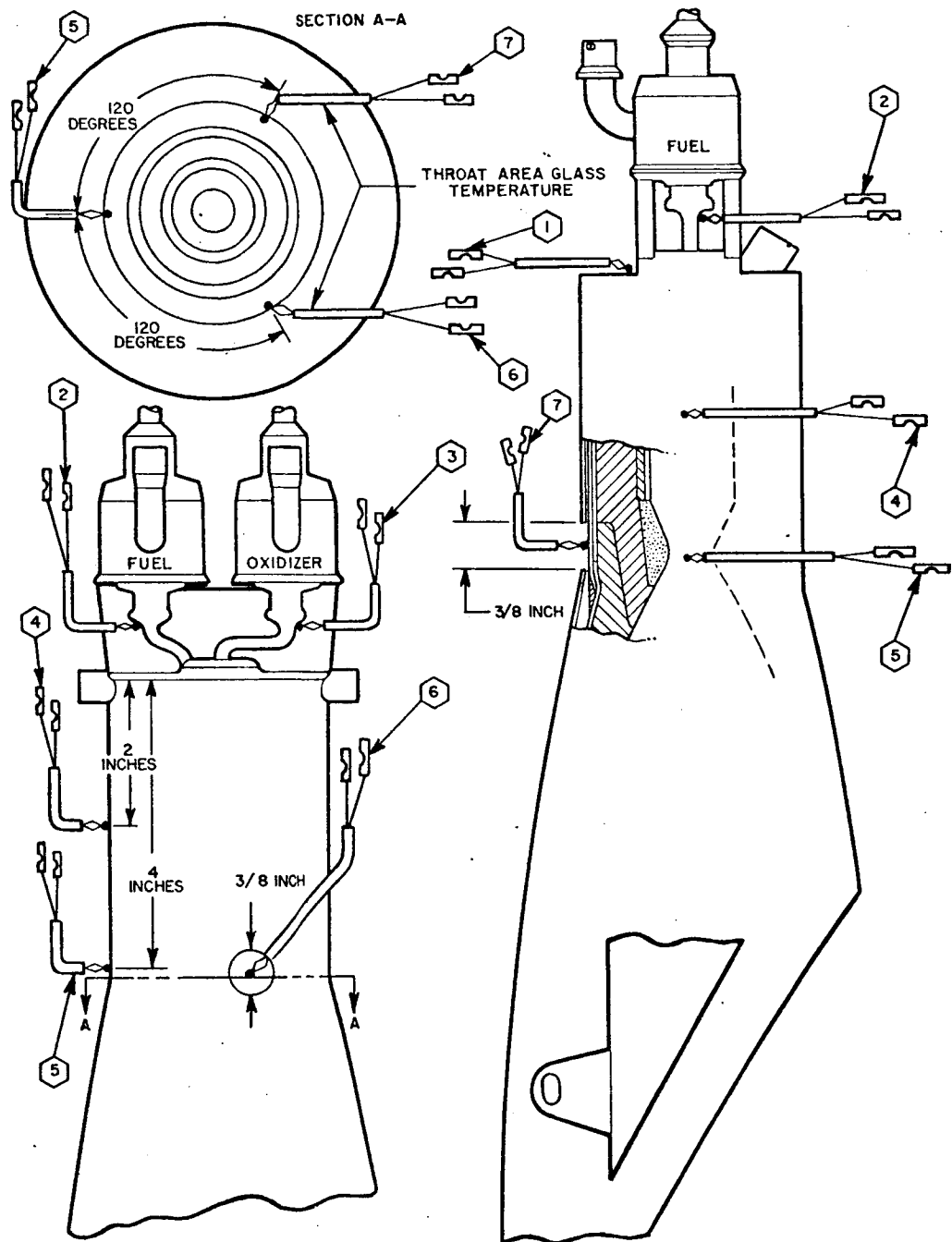
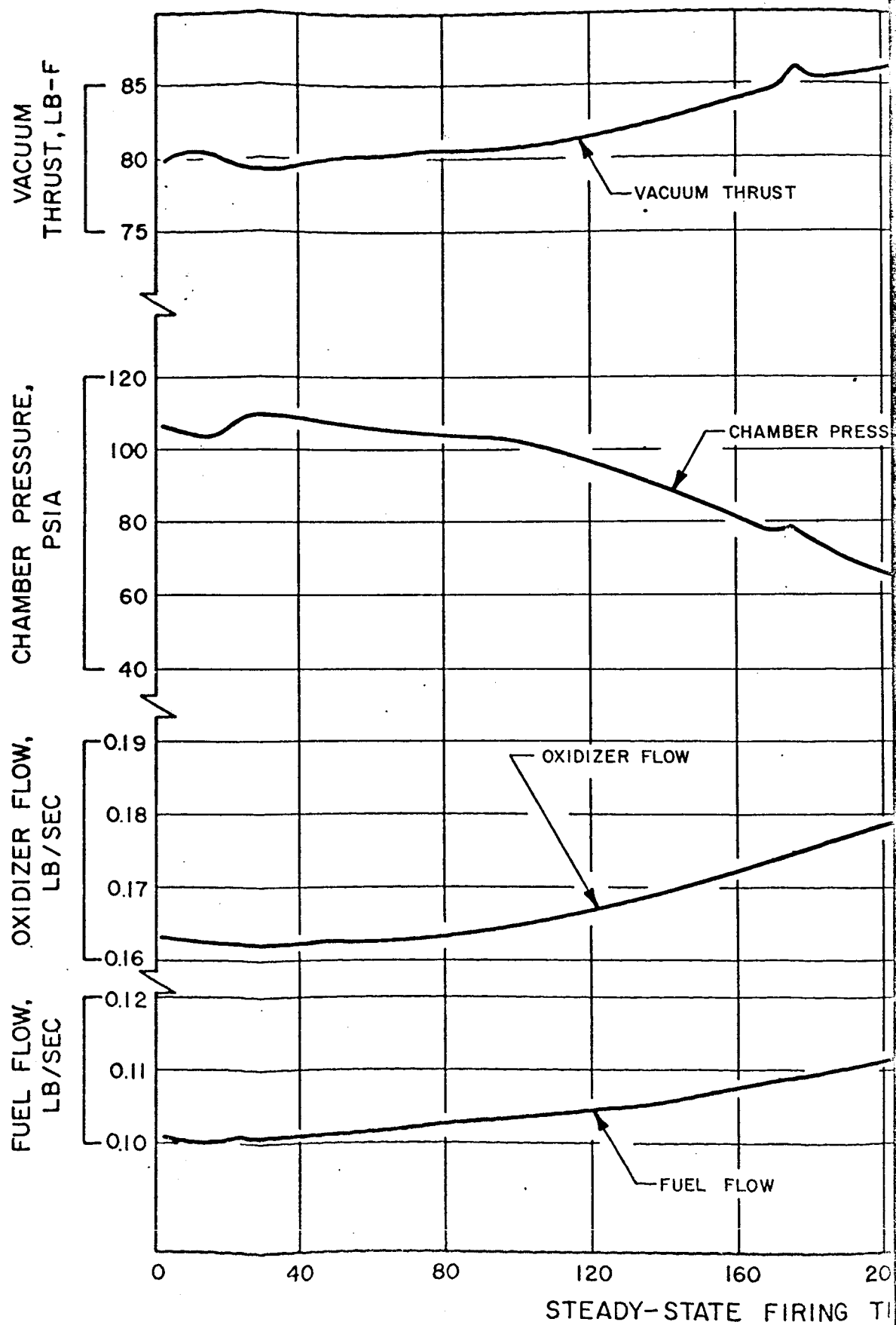


Figure 2. Gemini TCA Thermocouple Location Diagram

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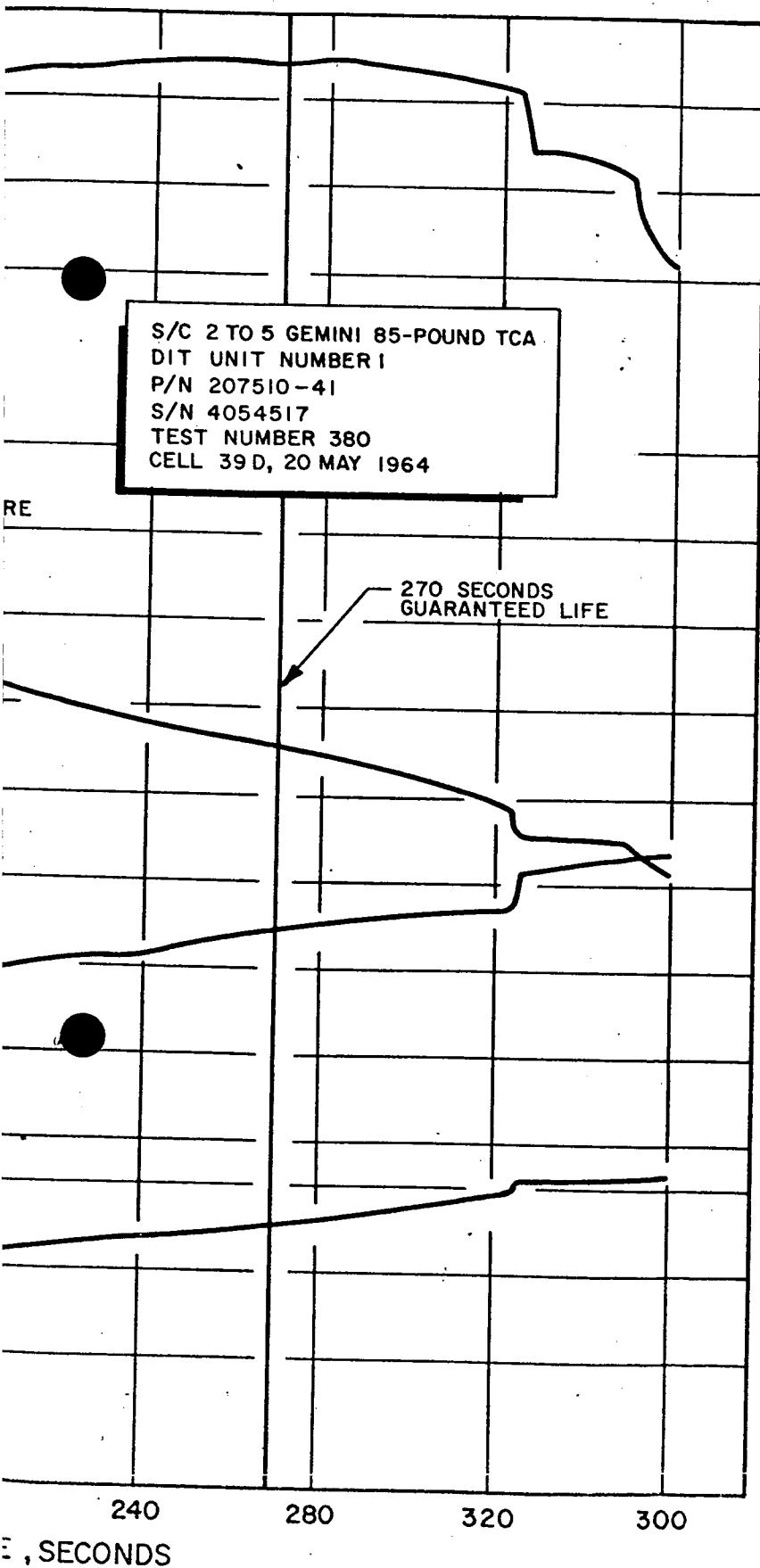


Figure 3. Continuous Firing Test,  
Gemini 85-Pound-Thrust  
TCA

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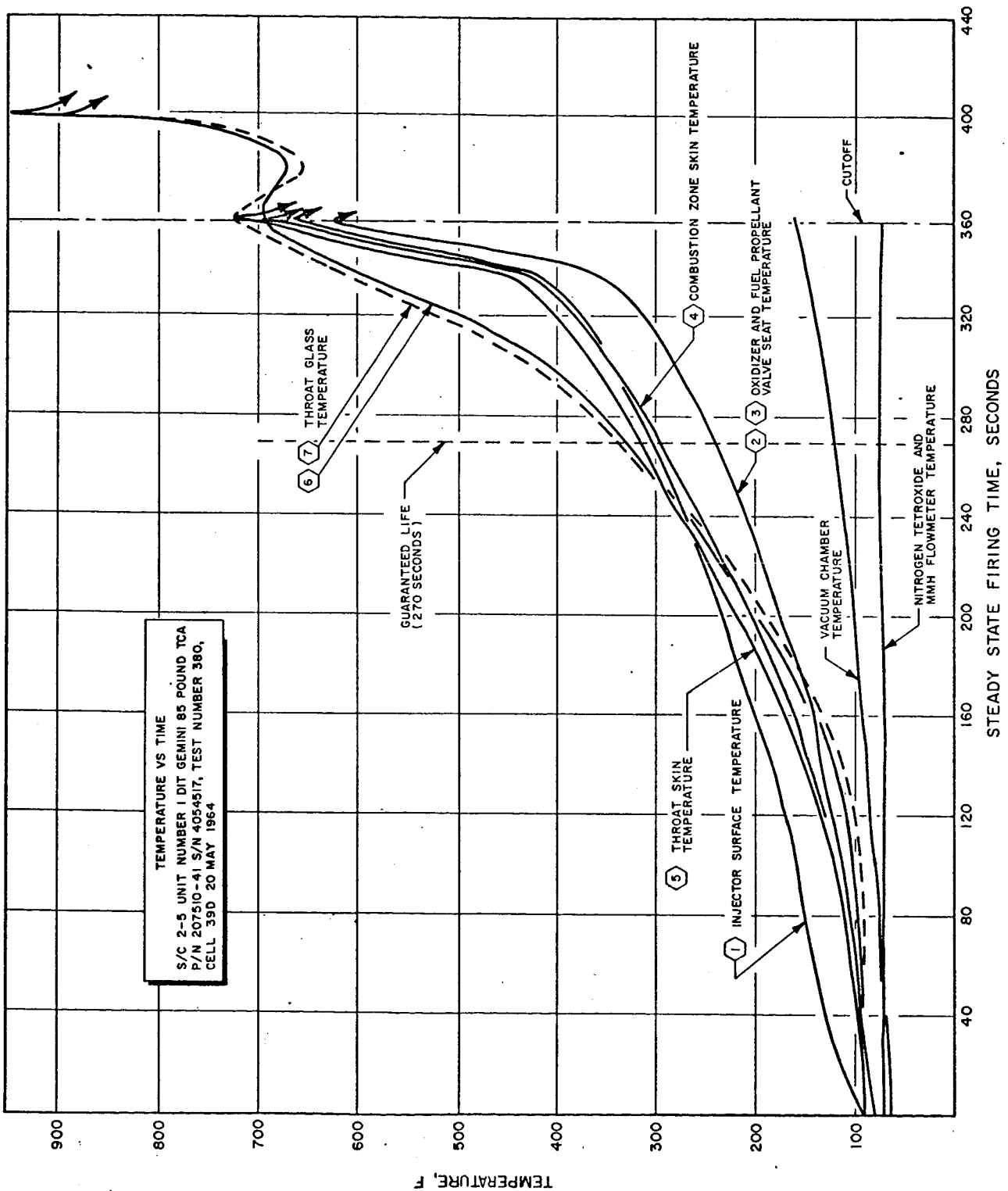
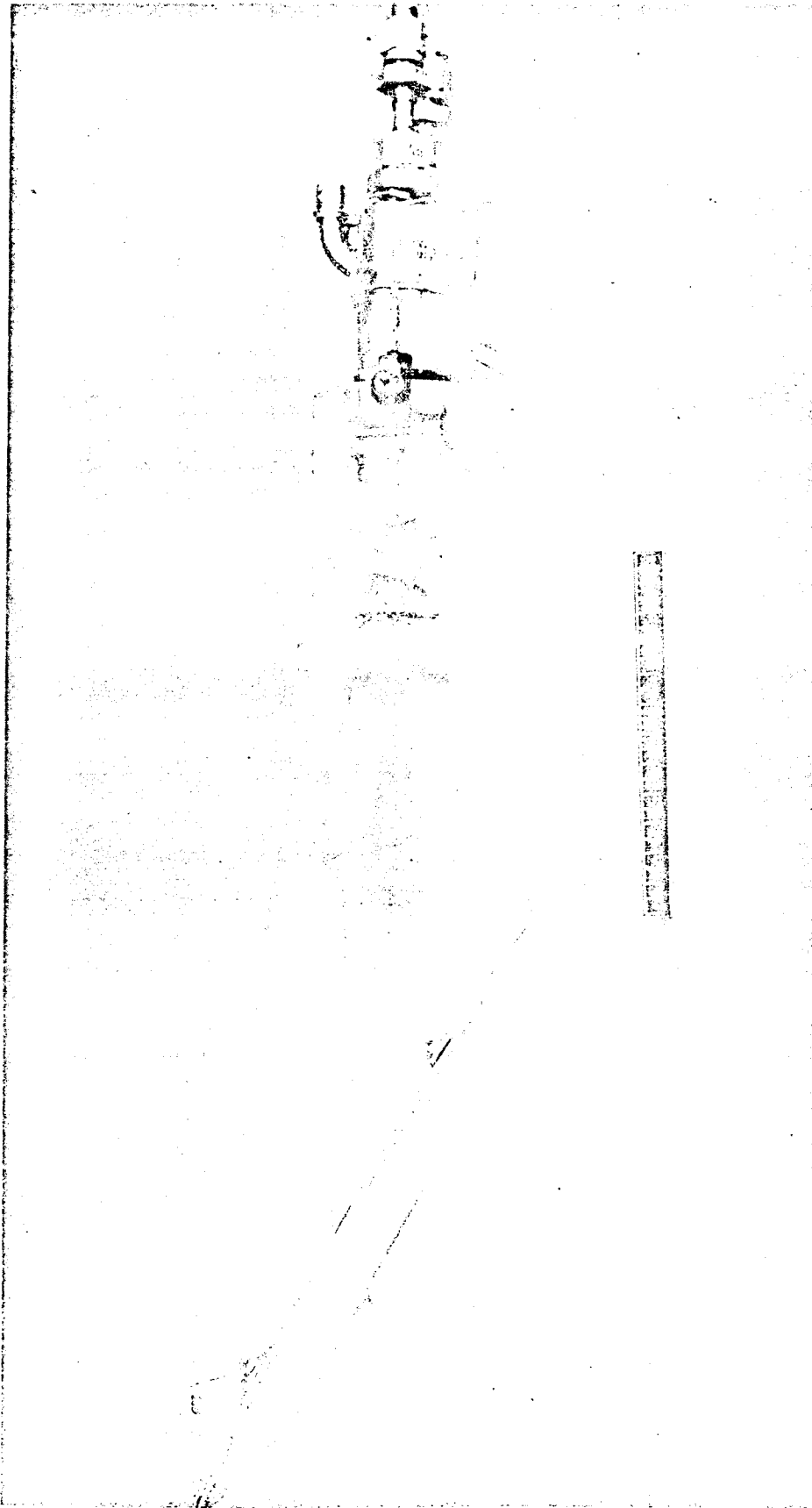


Figure 4. Temperature vs Time, Gemini 85-Pound-Thrust TCA

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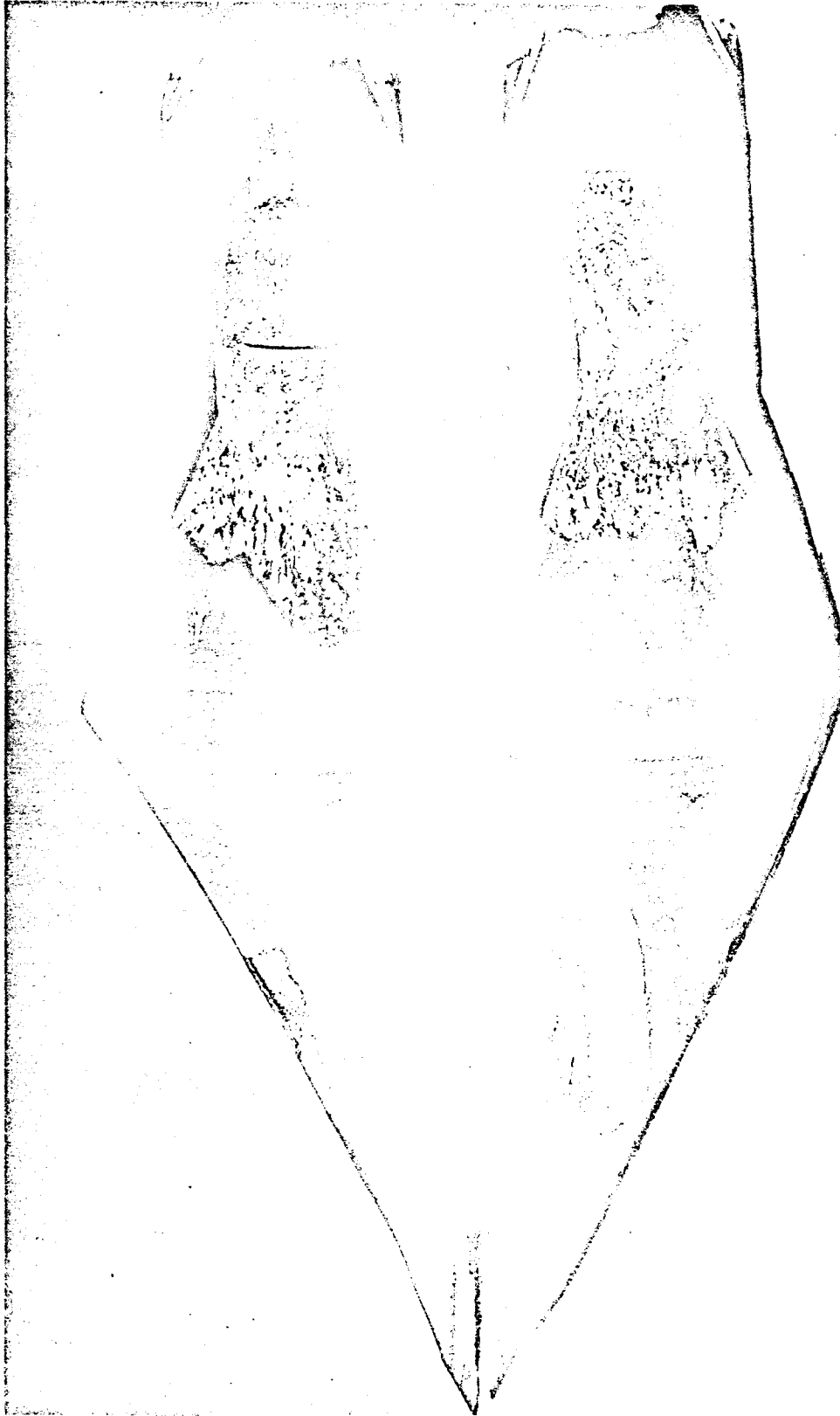
Figure 5. TCA Burnthrough Area

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Figure 6. Sectioned TCA

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APPENDIX B

CONTINUOUS TEST OF 100-POUND-THRUST  
TCA TO CHAMBER FAILURE

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TITLE: CONTINUOUS TEST TO CHAMBER FAILURE DESIGN INFORMATION TEST

INTRODUCTION

A continuous steady state firing until catastrophic failure of the thrust chamber was conducted on 18 May 1964, with a 100 lb flightweight thrust chamber, DIT Unit No. 1. The test was conducted at CTL-III, Cell 39D, at a simulated pressure altitude of 100,000 feet or greater and at local ambient temperature.

Catastrophic failure is defined as a condition in which further operation of the TCA would endanger the S/C or crew. It includes (1) loss of the capability to stop propellant flow; (2) flow or leakage of propellant or combustion products at locations in the TCA other than the TCA exit; and (3) TCA skin or mounting attachment temperature in excess of 600°F. All times in this report are from the start of steady state unless otherwise noted.

SUMMARY

The 100 lb flightweight thrust chamber was subjected to three calibration tests for a total of 9.6 seconds firing time, a cape cycle for an additional six seconds firing time, and a steady state continuous firing for 443.8 seconds, making a total accumulated firing time of 459.4 seconds. Catastrophic failure occurred after 388 seconds, when the injector surface temperature reached 600°F. The test was not terminated until a complete burn through occurred at the injector end. The burn through started after 425 seconds of firing time, but the test was continued for 34 seconds longer until test parameters clearly indicated failure.

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## DISCUSSION

### Chamber Configuration

The test chamber was a standard Capsule II production assembly P/N 207540-41, S/N 4057751, as shown in Figure 7. This thrust chamber had been a production unit until an improper installation of valve inlet tubes resulted in the thrust chamber being converted to an engineering test unit.

### Test Description

The thrust chamber was first calibrated to nominal thrust, mixture ratio, and inlet pressures with a total of three tests for 9.6 seconds of firing time. Following the calibration tests, a series of six one-second cape cycle tests were conducted at ambient conditions for 6.0 seconds firing time. The thrust chamber was instrumented with thermocouples (locations shown in Appendix A) in order to monitor skin and ablative temperature during the steady state test. The steady state continuous firing test was conducted at a simulated pressure altitude of 131,000 feet.

### Test Results

Calibration Tests - The performance values obtained during calibration testing were: Injector end  $C^*$  of 5359 ft/sec (91.7% theoretical shifting) at a mixture ratio of 1.61 and a chamber pressure of 133.2 psia obtained with valve inlet pressures of 270 psia for oxidizer and 227 psia for fuel.

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No attempt was made to change the propellant valve inlet orifice size to obtain design inlet pressures. This would have caused unnecessary delays to this program in view of the fact that propellant supply pressures could be adjusted to obtain nominal thrust and mixture ratio and the system dynamics have negligible effects on a steady state test. Vacuum  $I_{sp}$  was 16 seconds above nominal.

Cape Cycle - No data was obtained from the cape cycle tests because the one-second firing duration was not sufficient time for instrumentation to reach stable conditions.

Steady State Continuous Firing Test - The performance parameters at the start of the test were similar to the calibration tests. As the test progressed, thrust, chamber pressure, inlet pressures, and propellant flows changed as shown in Figures 8 and 9. Temperature of the TCA increased during the test as shown in Figure 10.

After 388 seconds of firing time the injector surface temperature reached 600°F, which is defined as catastrophic failure. The test was continued until a burn through occurred at the injector end of the TCA. The burn through started at 425 seconds, as evidenced by motion pictures of the test and increased until the test was terminated at 443.8 seconds. Total accumulated firing time was 459.4 seconds, including calibration, cape cycle, and steady state to failure.

Post-test inspection of the thrust chamber revealed that there were two burned through areas located 180° apart at the injector end of the TCA (Figures 11 and 13). One area was burned through the injector top surface and injector mounting fingers while the other area burned through the mounting fingers only. No other areas of the TCA shell displayed severe heating.

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The throat area was severely eroded with no trace of the throat insert remaining (Figure 12). The combustion zone liner and ablative sleeve were severely eroded with the most severe erosion in line with the injector fuel inlet.

Examination of the sectioned TCA (Figure 14) revealed that there was little virgin ablative material remaining in the TCA body. The ablative material in the region of the injector was badly charred, eroded, and delaminated with no trace of the "O"-ring seal or the adhesive bond between ablative and injector. The 0° wrapped ablative nozzle had some delaminations, and the 90° ablative material of the combustion chamber had several delaminations increasing in number adjacent to the injector.

The injector had suffered burned areas in the top surface and fingers as well as burned out portions of the splash plate. These burned out areas were in line with the fuel and oxidizer orifice pairs except for two adjacent pairs located 180° from the injector fuel inlet (Figures 13 and 17). The splash plate was eroded in line with these orifices but not burned completely away as was the case with the other 14 orifice locations.

#### ANALYSIS

A detailed analysis of the test data and thrust chamber assembly was made. It included an analysis of performance and temperature parameters during the firing period, a study of the TCA condition before and after the firing, and an investigation of the injector assembly geometry and flow characteristics.

Analysis of the data shows that the area of the throat decreased due to normal material buildup in the throat for the first 35 seconds of the steady state firing (see Figure 8). This is reflected in a lower thrust

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and propellant flow rate and a higher chamber pressure (up 12 psia) from the starting pressure of 134 psia.

The throat started eroding gradually from 75 seconds to 170 seconds at which time part of the throat insert was expelled through the nozzle. The chamber pressure had decayed to 105 psia while the thrust had increased to 105 lbs from the initial thrust of 96.16 lbs. During the next 50 seconds the throat insert again gradually eroded until at 318 seconds the remainder of the throat insert and liner relieved. The temperature of the TCA injector surface during the first 318 seconds of the firing appeared normal, but at this time it started to increase at a very rapid rate, Figure 10. Also, at this time, the thrust chamber pressure and flowrate of the propellant dropped which was not normal as flowrates usually increase when the chamber pressure decreases. One possible explanation of this phenomenon was that the propellant was being vaporized in the manifold of the abnormally hot injector. Injector surface temperature reached 600°F after 388 seconds of firing time. The increased temperature of the splash plate no doubt initiated accelerated propellant combustion within the splash plate area resulting in burning away portions of the splash plate in line with the orifice pairs. Loss of the splash plate resulted in a changed injector burning pattern which in turn eroded the injector end of the combustion chamber and eventually burned through the TCA ablative, shell, and injector.

Excessive injector temperature such as was encountered during the steady state continuous firing of this thrust chamber could result from hot gas leaks in the vicinity of the injector or an improper injector burning pattern.

An investigation of the manufacturing quality control records of fabrication of the TCA did not reveal any abnormalities. There were no cracks or separation between the injector and the thrust chamber ablative.

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Asbestos and glass wrap were normal with no separations. Although it is very difficult to determine from X-ray inspection if the "O"-ring seal was in place, there was some indication of an "O"-ring. TCA proof pressure tests did not reveal any leakage or structural weakness. Post-test examination of the chamber pressure injector boss did not reveal any sign of a leak in that region.

A detailed analysis of the injector from this TCA was made in which the injector was carefully removed from the TCA body and subjected to water flow. All orifices were found to be open with none plugged. Water flow through the fuel orifices (Figure 15) at design flowrate did not impinge upon the splash plate except at the locations where the splash plate was not burned away. Water flowing through the oxidizer orifices (Figure 15) at design flowrate did impinge at the edge of the burned area of the splash plate. When water was flowed through oxidizer and fuel orifices simultaneously (Figure 15) at design flowrates there resulted a shower type distribution rather than the normal half inch diameter collimated stream. This divergence of propellant no doubt contributed to erosion and injector end char of the ablative body.

The amount of water flowing through each oxidizer and fuel orifice was measured at total design flowrates. A plot of flowrate versus orifice location is presented in Figure 16. Mixture ratio for each pair of orifices was calculated for each of the 16 pairs and is also shown on Figure 16. The highest mixture ratio was 2.50 and 2.17 found at the orifice pair adjacent to the fuel inlet. This condition is normal for this type injector.

Following the water flow tests, the injector was sectioned (see splash plate, Figure 17) and measurements were made of the orifice diameters and impingement points. These measurements are presented in Figure 18.

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The post-firing orifice sizes are slightly smaller than pre-fire measurements due to material deposited in the orifices and distortion of the orifices because of excessive injector heat encountered at the time of burn through. The propellant streams impingement points were found to be out of tolerance also because of heat distortion of the injector. The inspection test buttons made during the manufacture of this injector indicated that the orifice drilling process was within specifications.

In order to be certain that orifice pair impingement points above the splash plate would not burn out a splash plate, a workhorse injector was fabricated with impingement points approximately 0.010 inch above the splash plates. This injector was fired in a workhorse TCA at design conditions for 440 seconds with no damage to the splash plate indicating that even if the impingement points were not in accordance with the specification the TCA should not have failed by first burning out the splash plate.

From the facts obtained from the precluding analysis it was hypothesized that the sequence of TCA failure was probably as follows:

1. The throat insert eroded severely in the area of high mixture ratio combustion gases.
2. Following the erosion, the throat insert cracked and was blown from the nozzle.
3. The absence of a throat exposed the ablative body of the TCA to combustion gases which started erosion of the ablative.
4. At the same time the exposed edges of the combustion chamber liner also were eroded away gradually, consuming all the liner segments.

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5. Coincident with the loss of segments the ablative in the combustion chamber area washed out in the high mixture ratio area.
6. The weakened ablative delaminated at the injector end and the hot gases began to leak around the injector, sharply increasing the injector temperature.
7. Increased injector temperature increased propellant burning inside the splash plate which burned away the splash plate in line with orifice pairs.
8. With the splash plate no longer functioning, the propellant spray diverged and quickly burned through the injector.

#### CONCLUSIONS

No definite conclusions could be drawn regarding the precise cause of the TCA failure. Investigation of the injector did not provide evidence that a defective injector propellant spray pattern initiated the erosion. However, the characteristic propellant distribution of this type of injector produces a high mixture ratio zone which erodes the TCA throat and combustion chamber during long duration tests. This erosion was evident from the beginning of this test and it was the most important cause of TCA deterioration.

Gemini S/C II through V 100 lb TCA's were not designed for steady state continuous firing and this type of operation will result in severe erosion and premature overheating and burn through of the thrust chamber.



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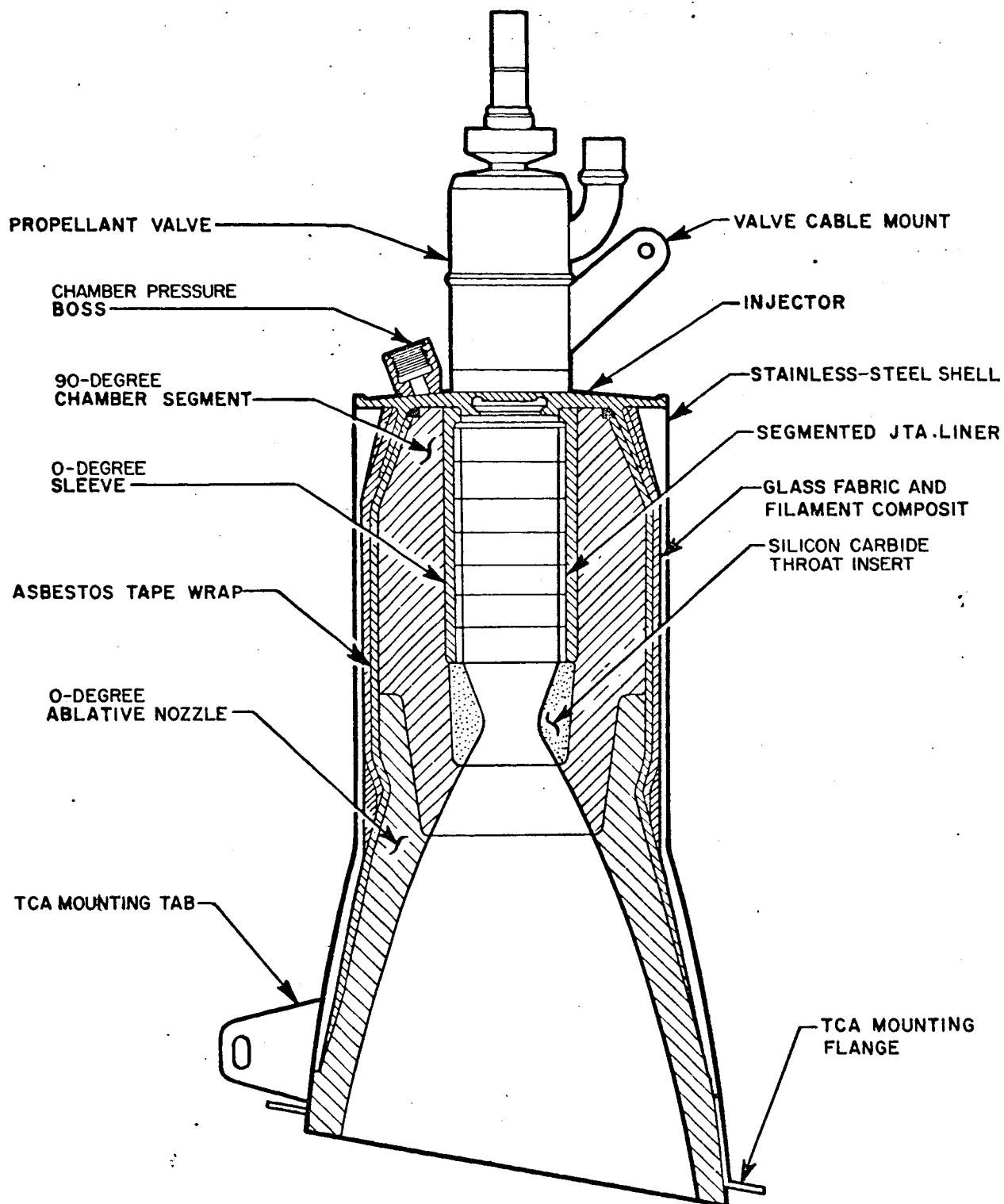
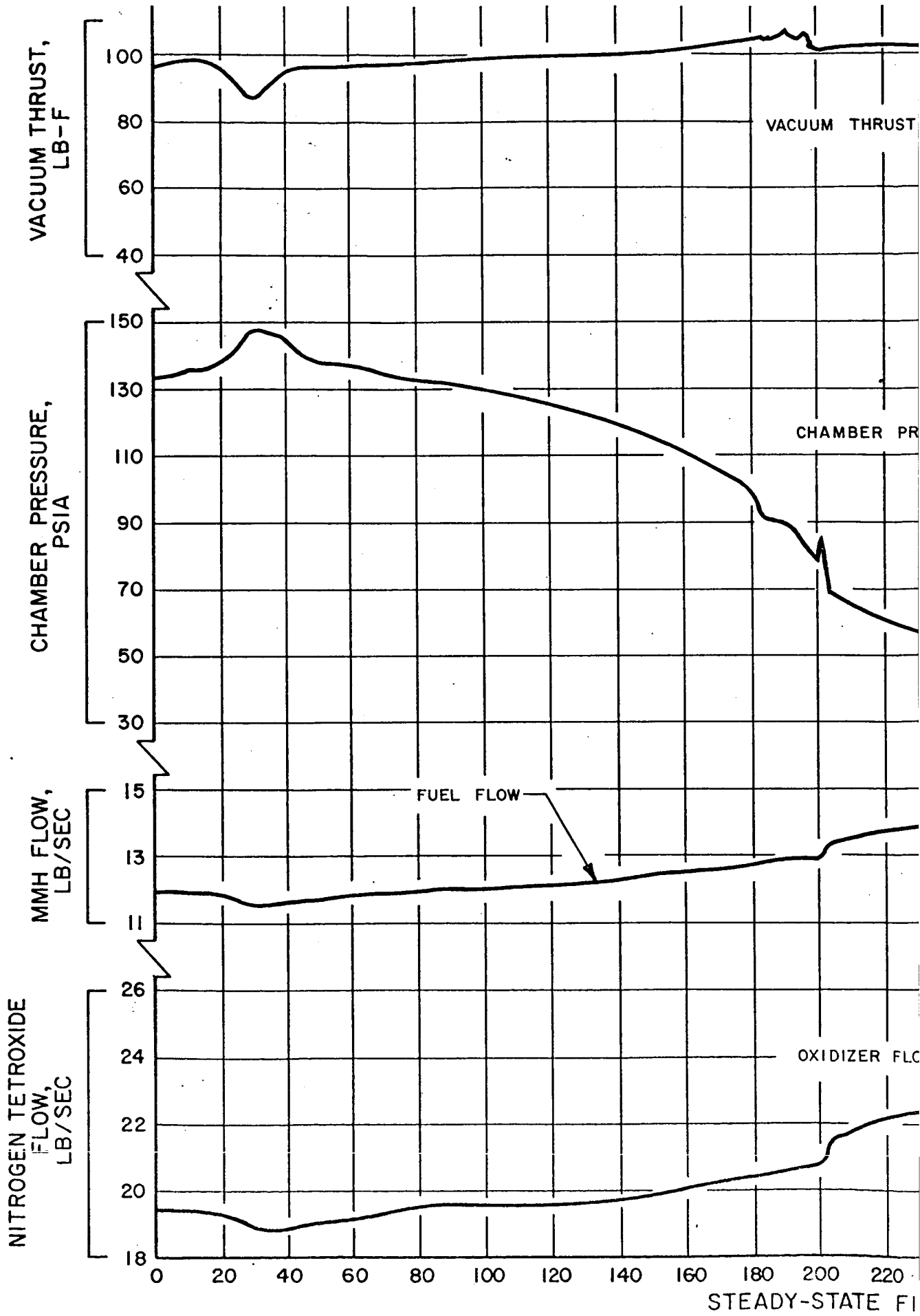


Figure 7. Gemini 100-Pound-Thrust TCA, Cross-Sectional Diagram

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S/C 2 TO 5 UNIT 1 DIT  
GEMINI 100- POUND TCA  
P/N 207540-41, S/N 4057751

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Figure 8. Continuous  
Firing Test, Gemini  
100-Pound-Thrust  
TCA

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260 280 300 320 340 360 380 400 420 440 460  
TIME, SECONDS



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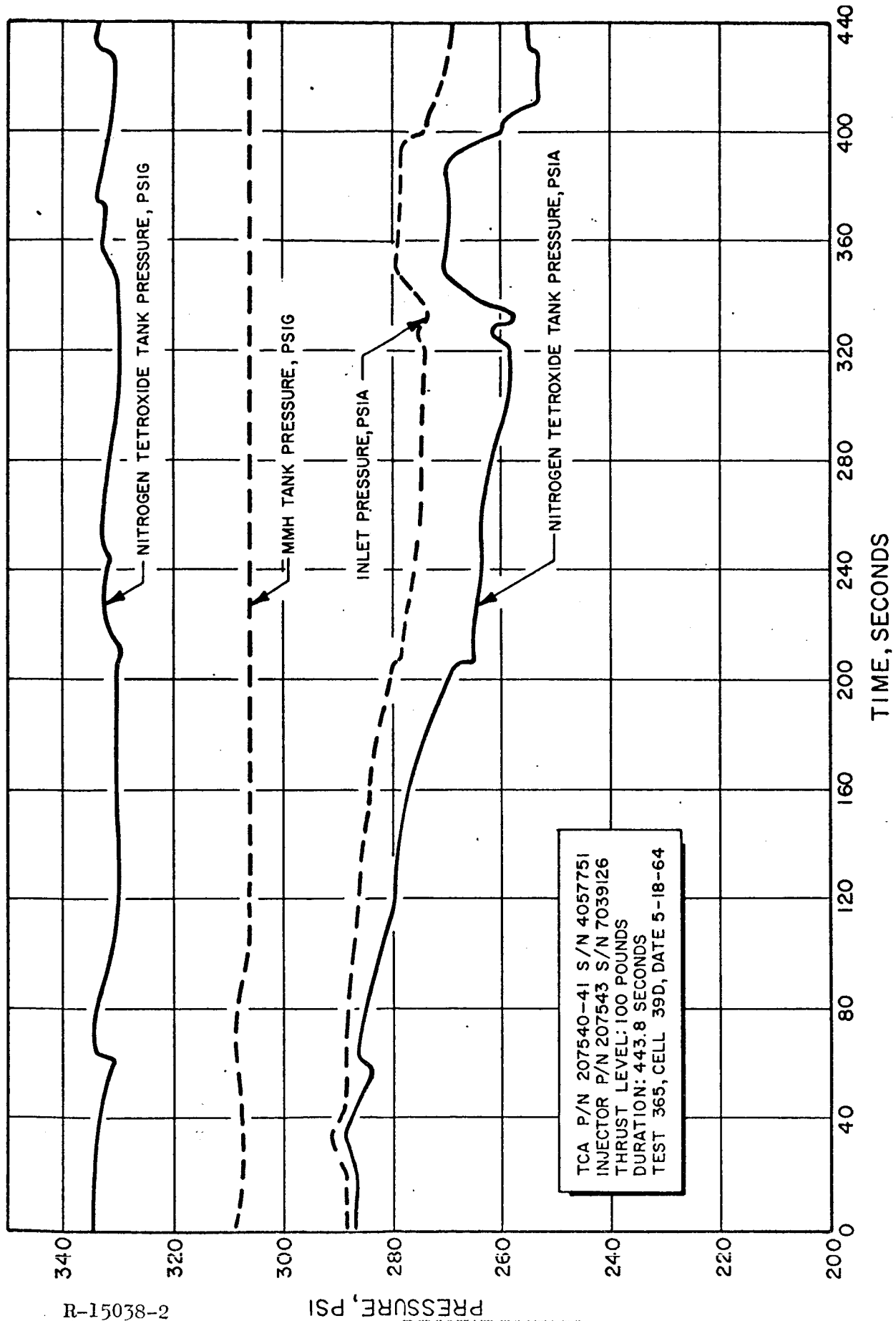


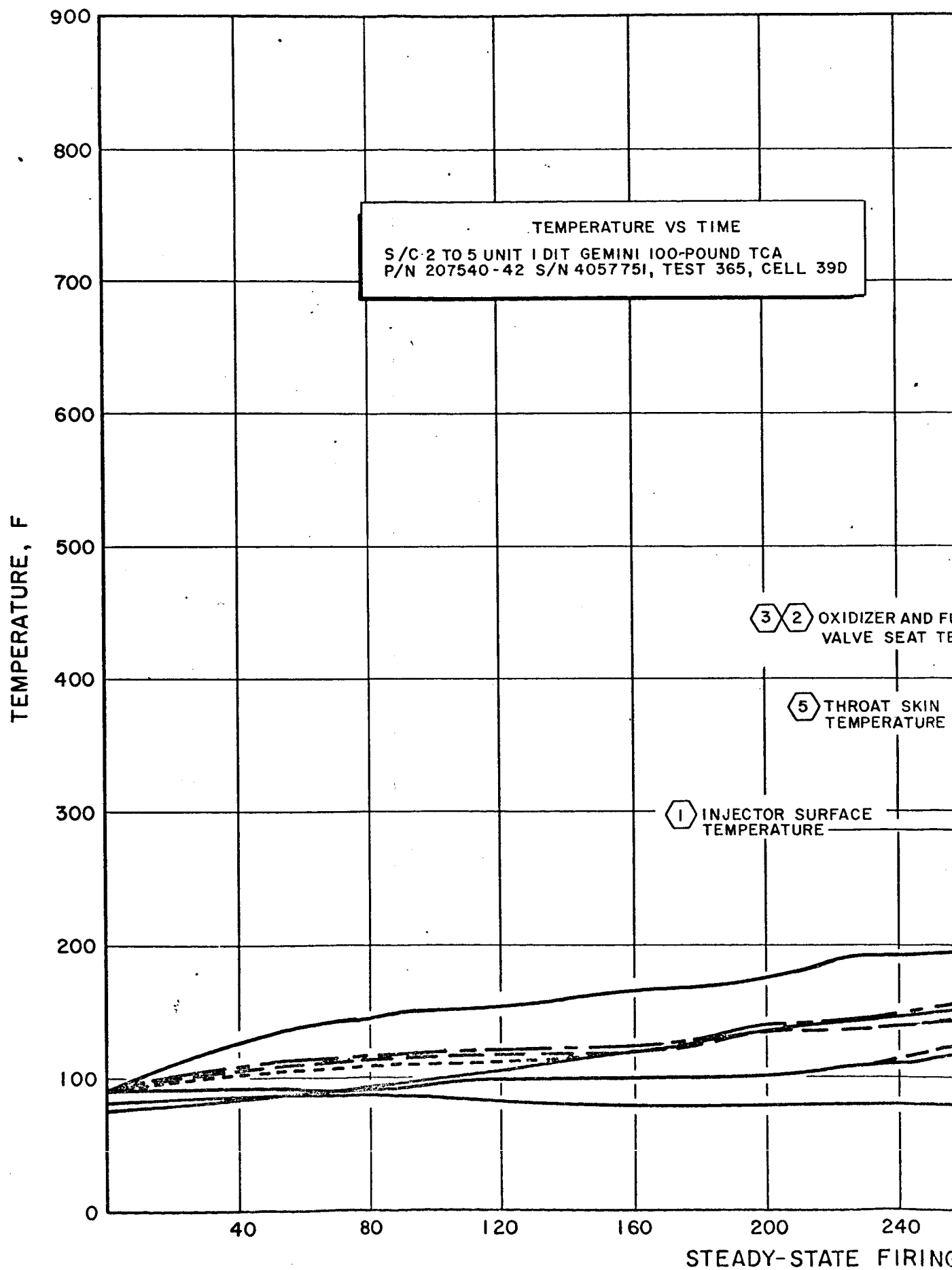
Figure 9. Pressure vs Time

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PRESSURE, PSI

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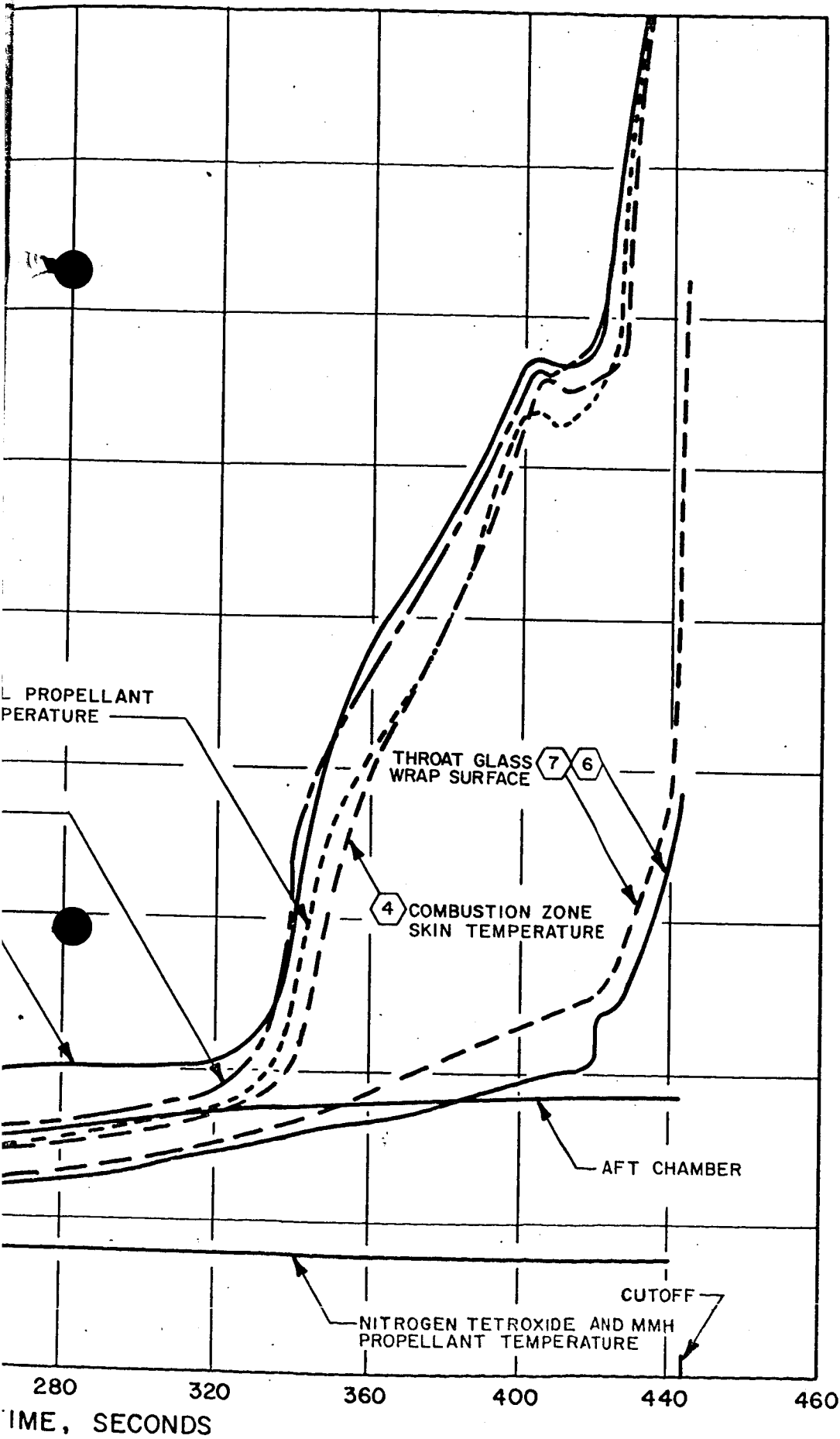


Figure 10. Temperature vs Time

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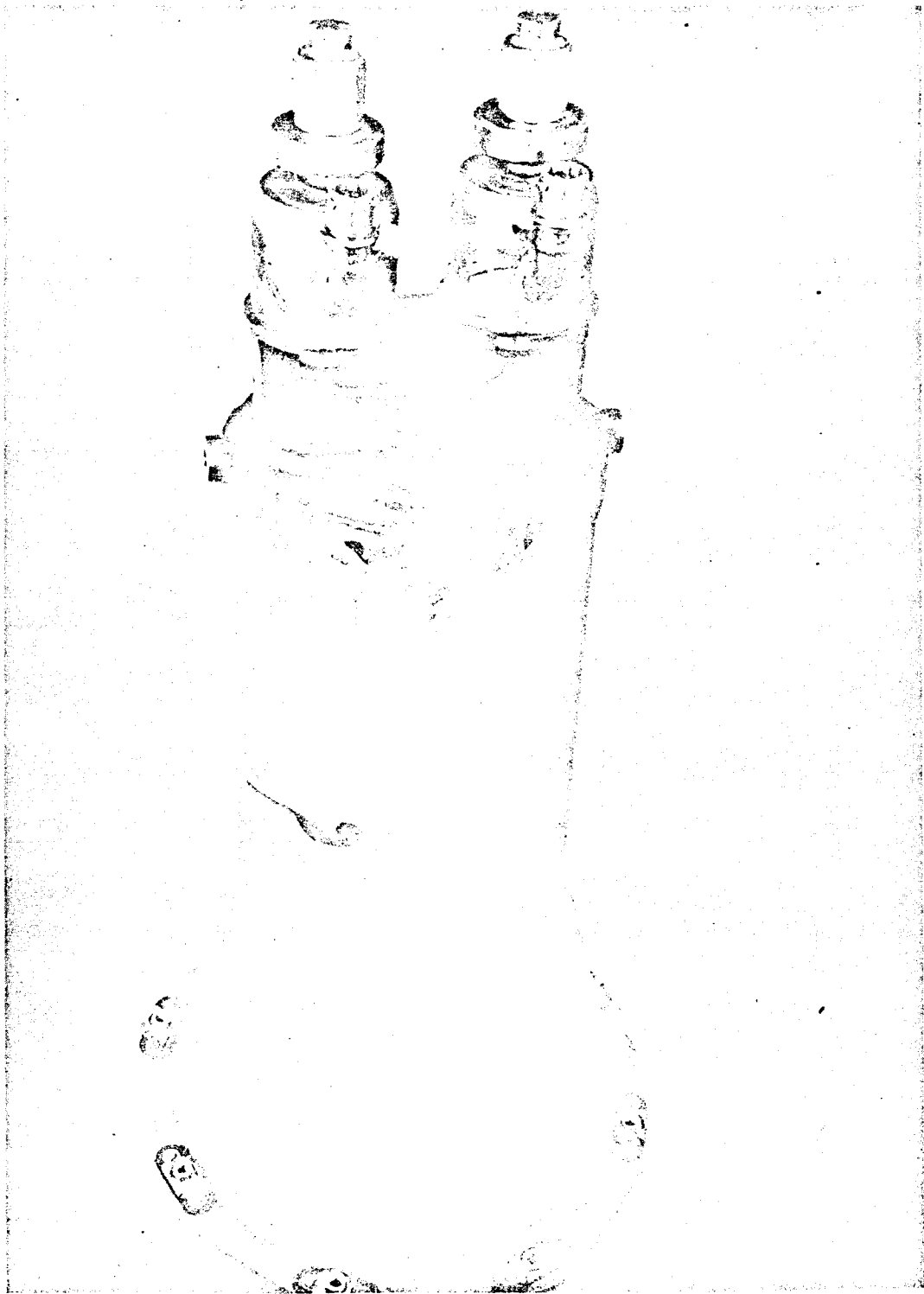
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Figure 11. Burnthrough Area

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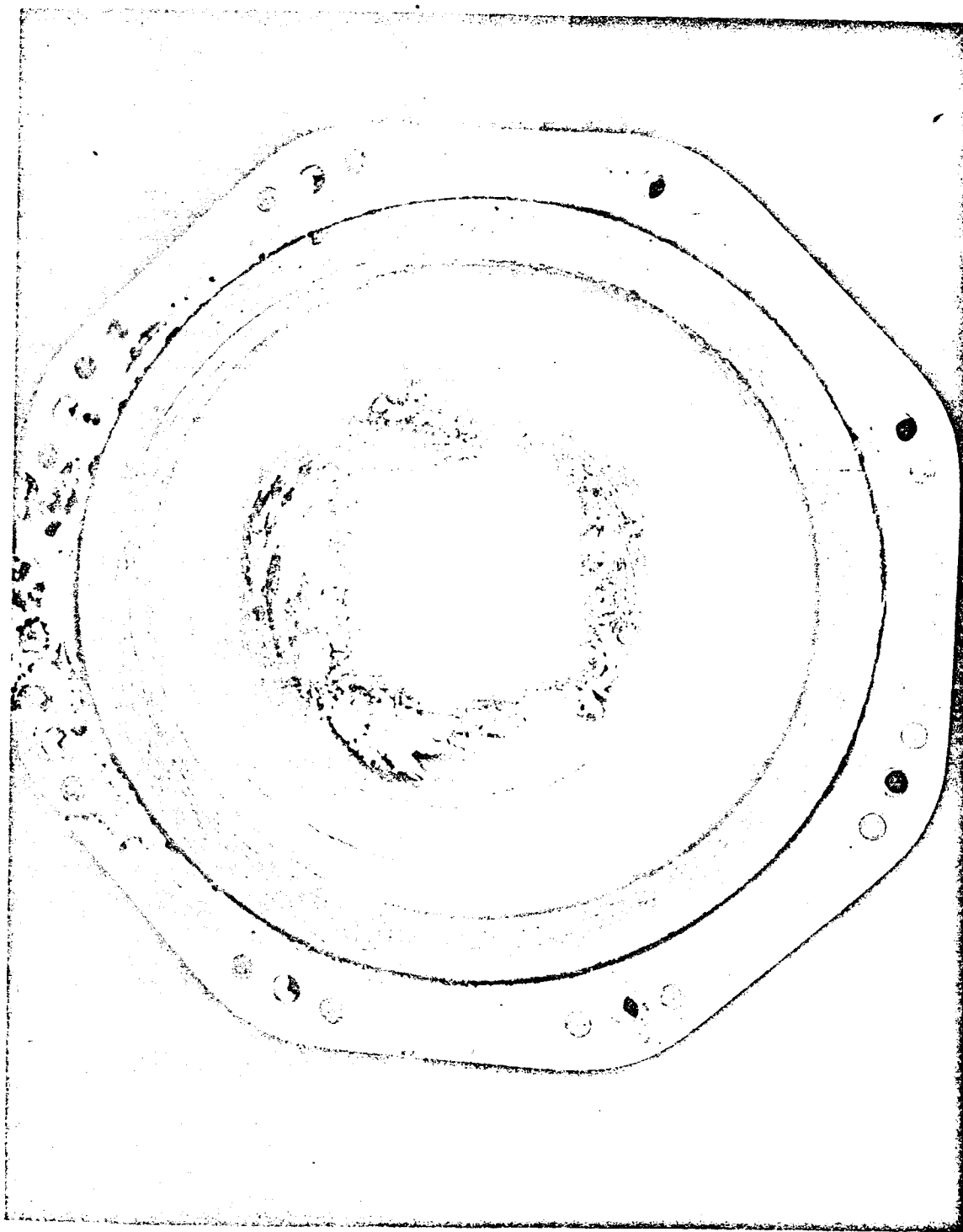
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Figure 12. Eroded Throat Area

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Figure 13. Injector-End Burnthrough Area

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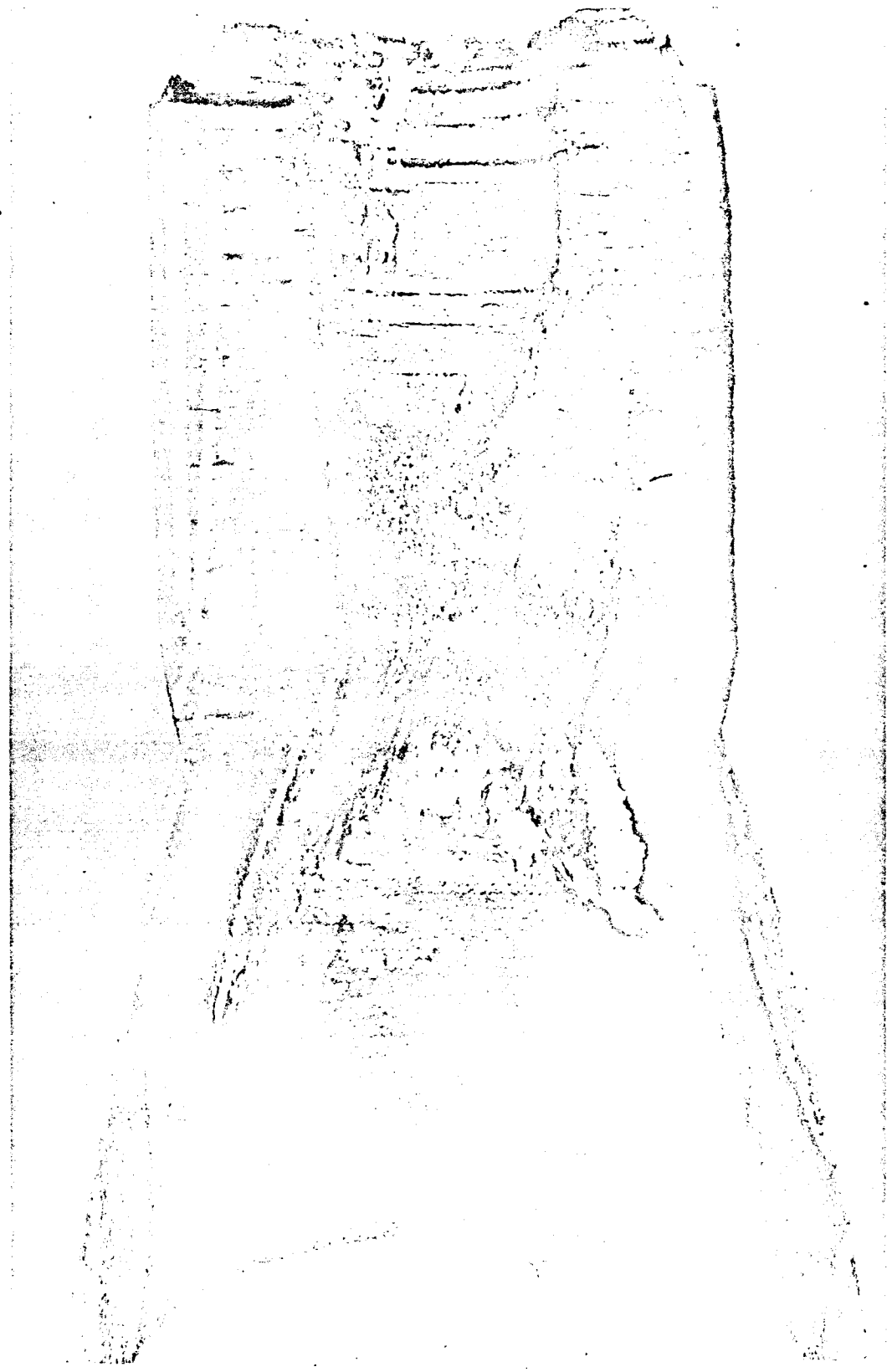
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Figure 14. Sectioned TCA

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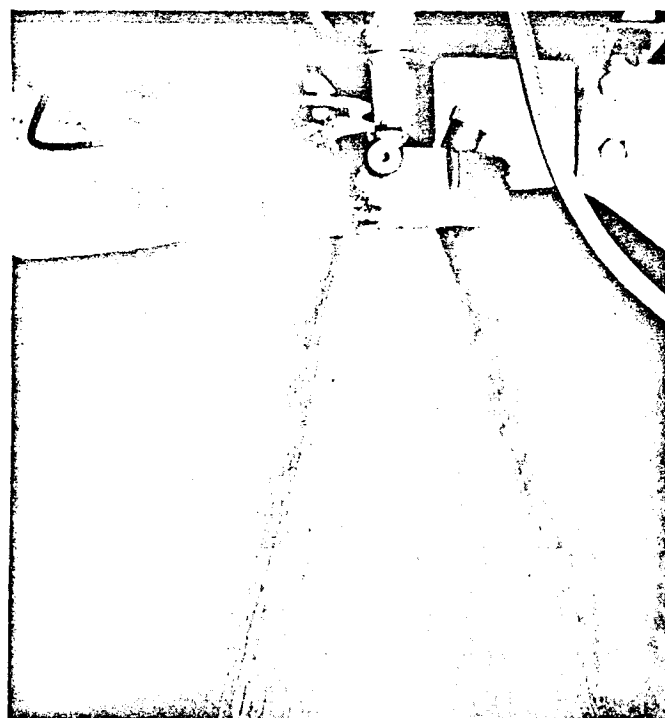


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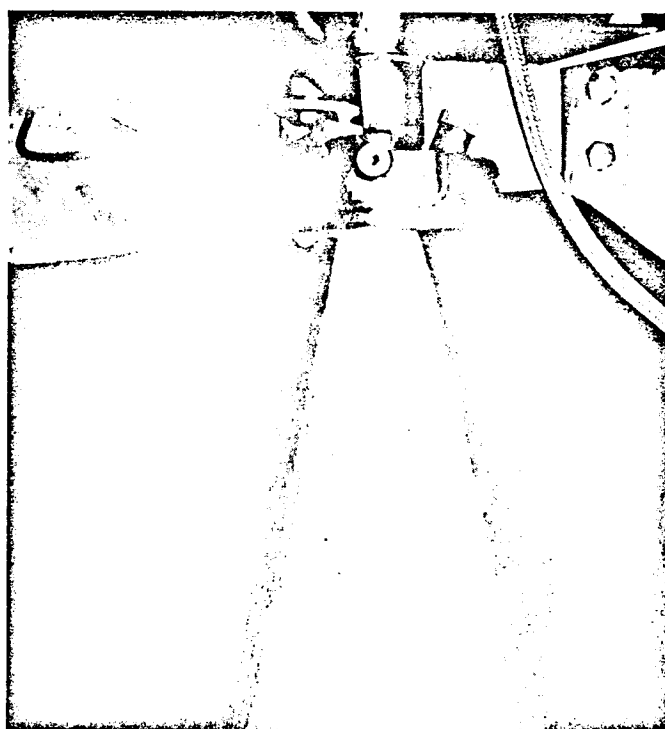
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(a) Fuel Only



(b) Oxidizer Only



(c) Oxidizer and Fuel

Figure 15. Flow Pattern With Splashplate

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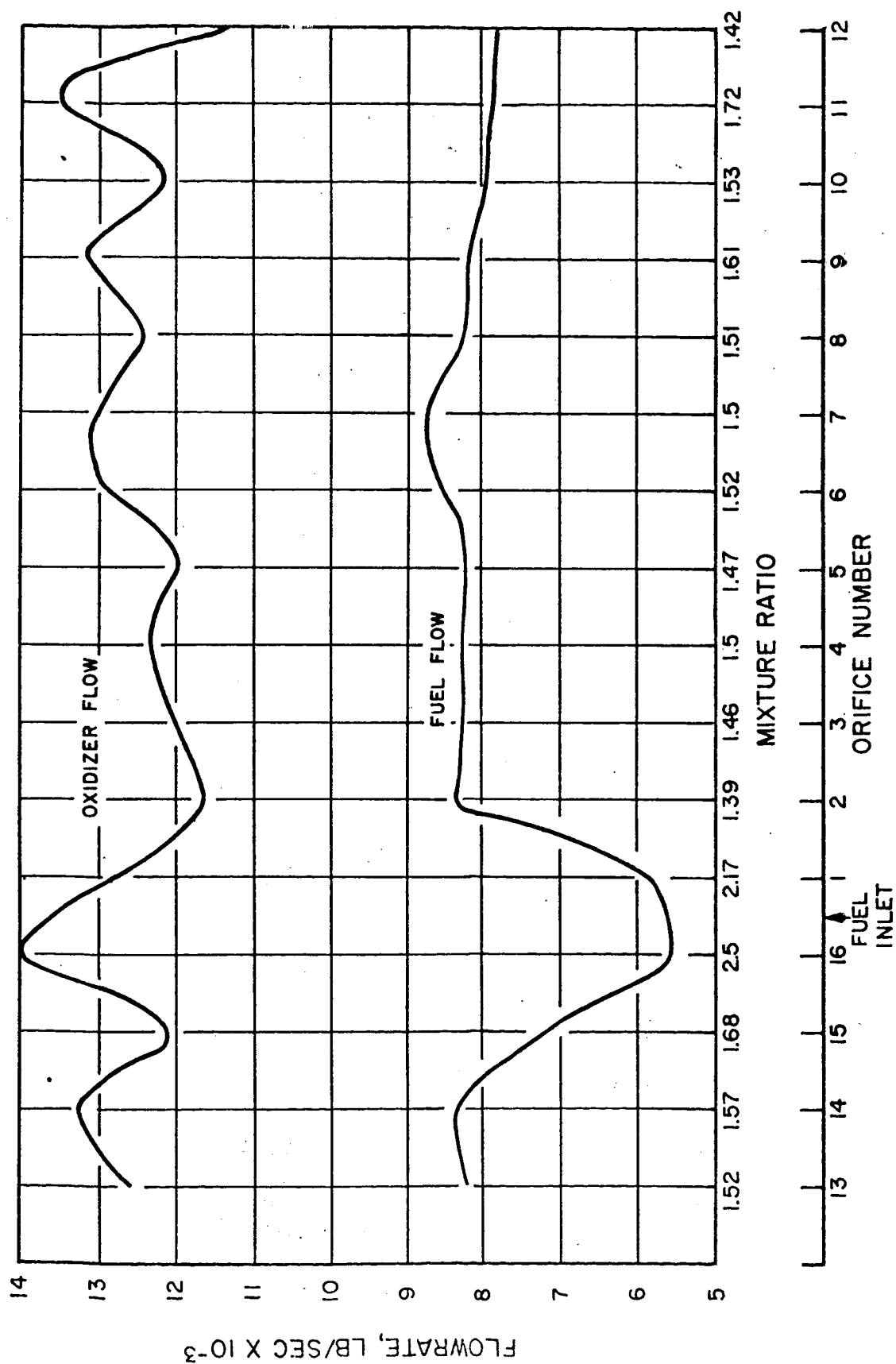


Figure 16. Fuel and Oxidizer Flow vs Orifice Number

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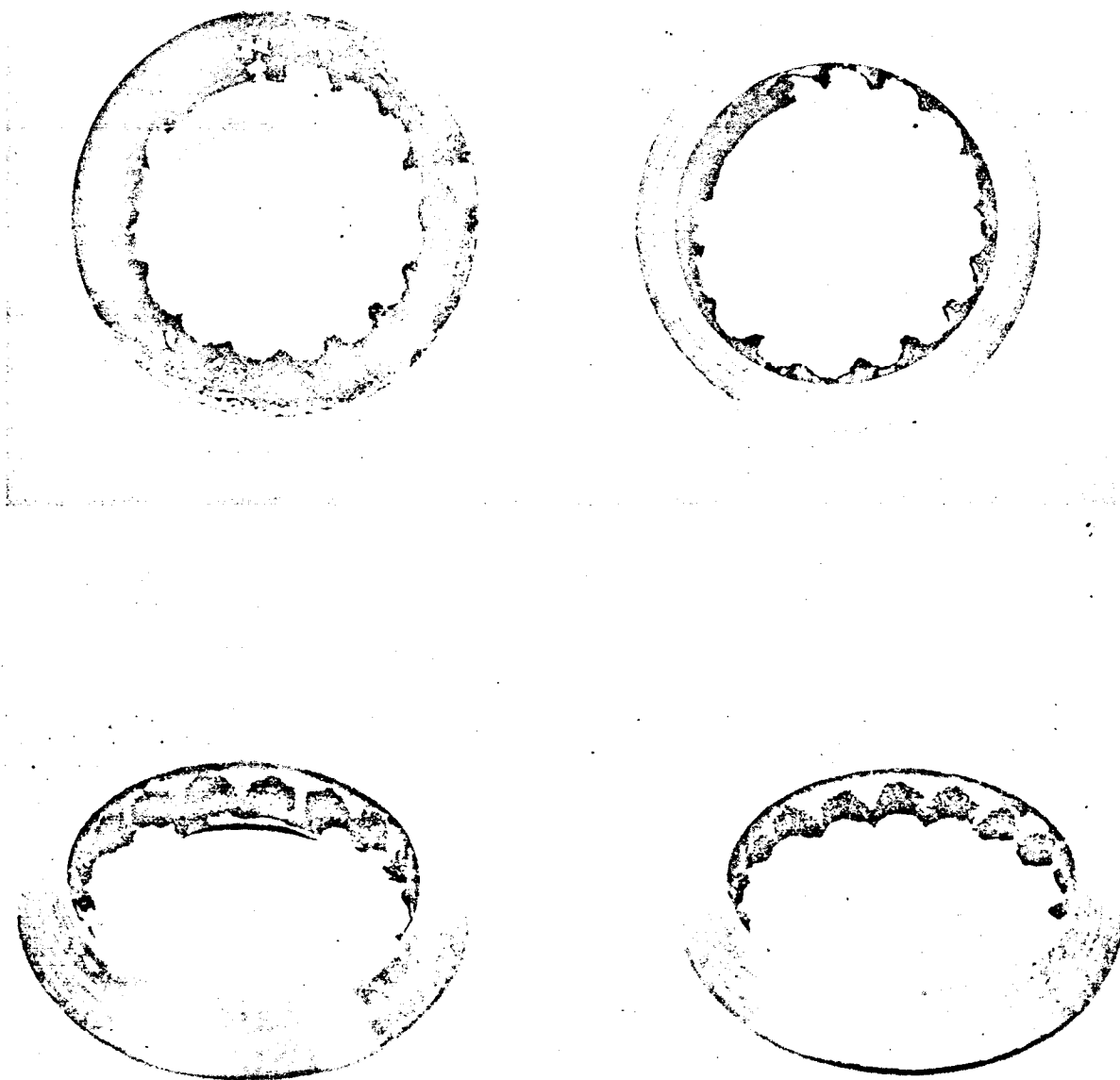
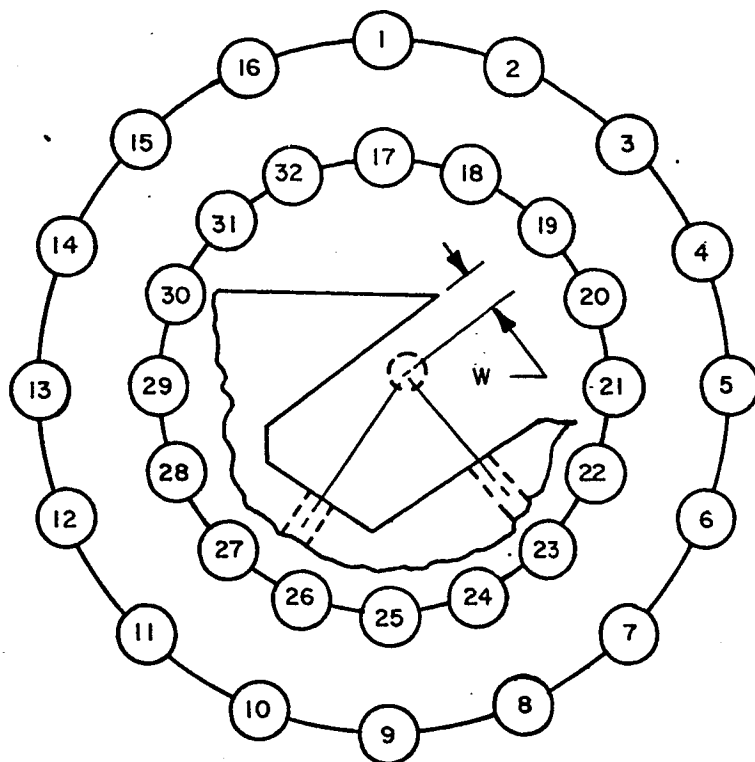


Figure 17. Splashplate

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MISS IMPINGEMENT

HOLE NO.	W
1 AND 17	0.0175
9 AND 25	0.014
11 AND 27	
4 AND 20	0.0135
12 AND 28	0.0117
5 AND 21	0.0142
13 AND 29	0.0162
6 AND 22	0.0123

P/N 207543  
S/N 3079391

PREFIRE	HOLE NO.	POSTFIRE
0.0225	1	0.0206
0.0225	2	0.0206
0.0225	3	0.0210
0.0225	4	0.0209
0.0225	5	0.0211
0.0225	6	0.0206
0.0223	7	0.0206
0.0225	8	0.0206
0.0224	9	0.0208
0.0224	10	0.0211
0.0225	11	0.0214
0.0224	12	0.0214
0.0223	13	0.0215
0.0223	14	0.0210
0.0224	15	0.0208
0.0225	16	0.0206

PREFIRE	HOLE NO.	POSTFIRE
0.0243	17	0.0235
0.0246	18	0.0234
0.0245	19	0.0234
0.0244	20	0.0230
0.0246	21	0.0230
0.0243	22	0.0234
0.0243	23	0.0233
0.0243	24	0.0233
0.0244	25	0.0233
0.0243	26	0.0234
0.0243	27	0.0236
0.0243	28	0.0231
0.0242	29	0.0231
0.0242	30	0.0231
0.0242	31	0.0228
0.0242	32	0.0228

Figure 18. Orifice Characteristics

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APPENDIX C

100-POUND-THRUST TCA LOW-TEMPERATURE TEST

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~~CONFIDENTIAL~~TITLE: EXTREMELY LOW TEMPERATURE DESIGN INFORMATION TESTINTRODUCTION

A series of extremely low temperature design information tests was completed on 3 August 1964, at PFL, CTL-III, Cell 39K, on a 100 lb Capsule II thrust chamber assembly. The TCA was subjected to a series of five (5) steady state duration firings of fifteen (15) seconds each after being soaked to a temperature of minus  $100 \pm 10^{\circ}\text{F}$ . The chamber was tested under conditions of local ambient pressure and was allowed to return to ambient temperature between tests before the cold soak was applied. Propellant valve temperatures were maintained at  $+32^{\circ}\text{F}$  or higher. The test objective was to determine the effect on start characteristics, performance, and life of the thrust chamber by operation after extended periods of cold soak.

SUMMARY

Five extremely low temperature design information tests were conducted under steady state durations of fifteen (15) seconds each. The chamber was first steady state calibrated to nominal thrust and mixture ratio. Engine performance for these tests was  $C^* = 5450 \text{ ft/sec}$ , with a shifting theoretical efficiency of 93% at a 1.6 mixture ratio. Because of the throat erosion during the cold tests and propellant temperature increases, the engine performance increased by two percent by the end of the testing. The erosion of the throat insert was in the area between two existing intersecting cracks. The response characteristics, as monitored by facility instrumentation, were identical for both the cold tests and calibration tests. No ill effects to engine performance or life can be attributed directly to the low temperature soak.

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## DISCUSSION

The test chamber was a Capsule II 100 lb flightweight thrust chamber P/N 207540-41, S/N 4057749. The chamber was designated for use as test only because of two longitudinal cracks in the throat insert and two cracks in the second liner segment. Calibration of the TCA was accomplished at Reno and verified at PFL, at a mixture ratio of 1.6, prior to the start of the low temperature tests.

To cool the TCA to the required temperature, a coil of quarter inch copper tubing was wound around the TCA shell and  $\text{LN}_2$  was flowed through the tubing until the temperature on the inside diameter of the throat insert stabilized at minus  $100 \pm 10^\circ\text{F}$ . A belt type heater blanket was wrapped around the propellant valves to maintain them at a temperature of plus  $32^\circ\text{F}$  or greater.

All testing was accomplished at CTL-III, Cell 39F, under conditions of local ambient pressure. Five steady state tests of fifteen (15) seconds duration were conducted. At the conclusion of each test, the thrust chamber was allowed to return to ambient temperature and inspected before the cold soak was again applied. Overall engine performance was unaffected as a result of the cold mode tests. Increases in propellant temperatures, up to  $40^\circ\text{F}$ , during the individual tests caused the flowrates to decrease. This change in flowrate is attributed to changes in propellant densities, as much as  $1.5 \text{ lb/ft}^3$ , which was brought about by the exhaust gases re-circulating up past the thrust chamber causing an increase in propellant temperature. Thrust chamber performance, therefore, had a tendency to increase during the test runs. Analysis of the oscillographs showed some blockage of the throat for 130 and 200 milliseconds at the start of the two of the cold soak tests. This can be attributed to material buildup or scale blocking the throat. Chamber pressure then returned to normal.

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On the last two tests, throat erosion was evident and a decrease in chamber pressure resulted. The erosion was in the area between the two intersecting cracks in the diverging portion of the throat insert. All start and shutdown valve responses were identical for both the calibration tests and the cold tests.

Post-test inspection revealed no additional damage to the liner segments, but another longitudinal throat crack was present. Also a horizontal throat crack just forward of the minor diameter. Erosion of the throat was clearly visible in this area. X-rays revealed no damage to the ablative chamber. Total engine burn time was 92.5 seconds.

#### CONCLUSIONS

The results of cold mode testing shows no major effects on thrust chamber performance or start characteristics. TCA performance increases are attributed to poor evacuation of the hot exhaust gases around the test cell, which caused increases in propellant temperatures. This problem has been remedied to prevent re-occurrence. The erosion sustained by the throat insert is attributed to the condition of the insert prior to testing. The erosion was in the area between two existing intersection cracks in the diverging section of the insert. No additional damage was sustained by the liner segments, and the ablative chamber was in excellent condition. Analysis reveals no evidence that cold mode testing has any ill effect on TCA performance or life.

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APPENDIX D

OFF MIXTURE RATIO

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TITLE: OFF MIXTURE RATIO, DESIGN INFORMATION TEST

- Enclosures:
- (1) Figure 19, 100 lb TCA Cross Section Diagram
  - (2) Figure 20, Temperature vs Time
  - (3) Figure 21, Characteristic Velocity vs Mixture Ratio
  - (4) Figure 22, Specific Impulse vs Mixture Ratio
  - (5) Figure 23, Thrust Coefficient vs Mixture Ratio
  - (6) Figure 24, Post-Firing TCA Photograph

INTRODUCTION:

On 23 and 24 July 1964, a Spacecraft II thrust chamber was subjected to a series of twelve steady state tests of 3.2 seconds duration each and an Aft Firing mission duty cycle test to guaranteed life conducted off-design mixture ratio at simulated pressure altitude above 100,000 feet.

The test objective was to determine the effects of off-design mixture ratio operation of TCA performance and life. This constituted shifting the propellant mixture ratio ten and fifty percent above and below design nominal.

The thrust chamber selected for this test program had two longitudinal hairline cracks in the throat insert.

SUMMARY

The twelve performance evaluation tests which constituted shifting the propellant mixture ratio ten and fifty percent above and below design nominal, were successfully completed.

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Post-test inspection showed no visible change in the throat insert on completion of the above test series.

The off-mixture ratio ( $MR = 2.4$ ) Aft Firing mission duty cycle test did not meet guaranteed life. The thrust chamber exterior shell temperature exceeded the SCD requirement of  $600^{\circ}F$ , which is defined as catastrophic failure. The maximum temperature of the TCA was at the throat area and registered  $750^{\circ}F$  at cutoff. In addition to the excessive TCA exterior shell temperature, there was a decay in chamber pressure and thrust. Post-test inspection showed the segmented liners, zero degree wrap sleeve, throat insert, and adjacent  $90^{\circ}$  ablative in the combustion chamber were eroded away.

Total accumulated burn time was 422.5 seconds, which included twelve 3.2 second off-mixture ratio performance tests.

Upon sectioning the TCA, the ablative was found to be completely charred, but not severely delaminated.

## DISCUSSION

### Chamber Configuration

The thrust chamber used for this test was an S/C 2 100 lb flightweight TCA P/N 207540-41, S/N 4053226. It employed a seven (7) segmented JTA refractory liner,  $90^{\circ}$  chamber segment, zero degree wrap sleeve, a solid SiC throat insert, and a zero degree wrap exit nozzle, as shown in Figure 19. In addition, seven (7) Iron-Constantan thermocouples were installed on the thrust chamber at selected radial and axial chamber stations as indicated on diagram of the temperature profile (reference Figure 20).

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This TCA was selected for DIT because of two longitudinal hairline cracks in the throat insert.

#### Test Description

The thrust chamber assembly was installed in Cell 39E, at CTL-III, and mounted in a vertical firing down position. A heat shield was installed on the thrust chamber scarfed nozzle, in order to more closely simulate actual capsule conditions.

Test efforts were directed to the following objectives:

1. To evaluate TCA performance (thrust,  $C^*$ ,  $C_f$ , and  $I_{sp}$ ) as a function of shifting the propellant mixture ratio to ten and fifty percent above and below design nominal.
2. To determine off design mixture ratio TCA life capabilities with OCM-218 MDC at  $MR = 2.4$ .

#### Test Results

1. The evaluation of TCA performance (thrust,  $C^*$ ,  $C_f$ , and  $I_{sp}$  as a function at a varied mixture ratio) was successfully accomplished. Post-test inspection showed no visible change in the TCA condition.
2. Thrust chamber guaranteed life with OCM-218 MDC firing at off-design mixture ratio was completed. However, it is considered a failure due to thrust chamber exterior shell temperature exceeding SCD requirements of 600°F between the 5th and 6th pulse of the MDC. All OCM-218 MDC test temperatures recorded and their locations are graphically plotted and identified in Figure 20.

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In addition to the excessive TCA exterior shell temperature, there was a decay in chamber pressure and thrust, due to severe erosion in the combustion zone and total throat erosion. This decay started after approximately 10 seconds of the 15 second separation pulse and continued gradually throughout the MDC.

### TEST ANALYSIS

#### Performance

The data derived during the twelve 3.2 second duration test series, which constituted shifting the propellant mixture ratio ten and fifty percent above and below design nominal is graphically plotted (Figures 21 - 23) and show the thrust chamber characteristics at these different levels.

These test data demonstrate that the 100 lb S/C 2 thrust chamber was performing as follows during the MDC: 92.6% of shifting theoretical  $C^*$ , with an average vacuum thrust of 89.6 lbs at 2.4 off-design mixture ratio. The vacuum  $I_{sp}$  and  $C_f$  was 297.6 seconds and 1.8299, respectively.

#### Post-Test Hardware Condition and Analysis

Post-test inspection of the thrust chamber revealed that there was no burn through.

After sectioning the thrust chamber along the longitudinal axis into two halves (Figure 24), inspection revealed the throat insert was completely eroded away, as was the combustion chamber sleeve and segmented liners, causing the inner wall of the combustion zone to be exposed to extremely

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hot gases, thus causing high exterior shell temperatures. Some of the silica reinforcing material in the ablative body and sleeve was melted and deposited in the nozzle of the TCA.

Sectioning of the 100 lb S/C 2 thrust chamber substantiated that failure of temperature was caused by nearly total depletion of the pyrolyzing ablative material in the combustion zone.

The sectioned thrust chamber was found completely charred as indicated by the white dotted line in Figure 24. The charred material on the other hand, is extremely strong and not severely delaminated.

After approximately 10 seconds of operation into the separation pulse of the off-mixture ratio OCM-218 MDC, gradual erosion began and pressure decrease continued at a gradual rate until cutoff, indicating a steady increase in burnout of the throat insert and combustion zone area. As firing continued, the temperature exceeded 600°F on the TCA exterior shell. This thrust chamber failed in the expected manner for off-design mixture ratio operation.

Total accumulated burn time was 422.5 seconds, which included twelve 3.2 second off-mixture ratio performance tests.

#### CONCLUSIONS

The OCM-218 MDC guaranteed life firing at 2.4 off-design mixture ratio served to verify that operational capability at this level is possible, but not recommended with existing TCA configuration. Under these conditions, the TCA will not meet SCD requirements because of temperature requirements and loss in thrust and specific impulse.

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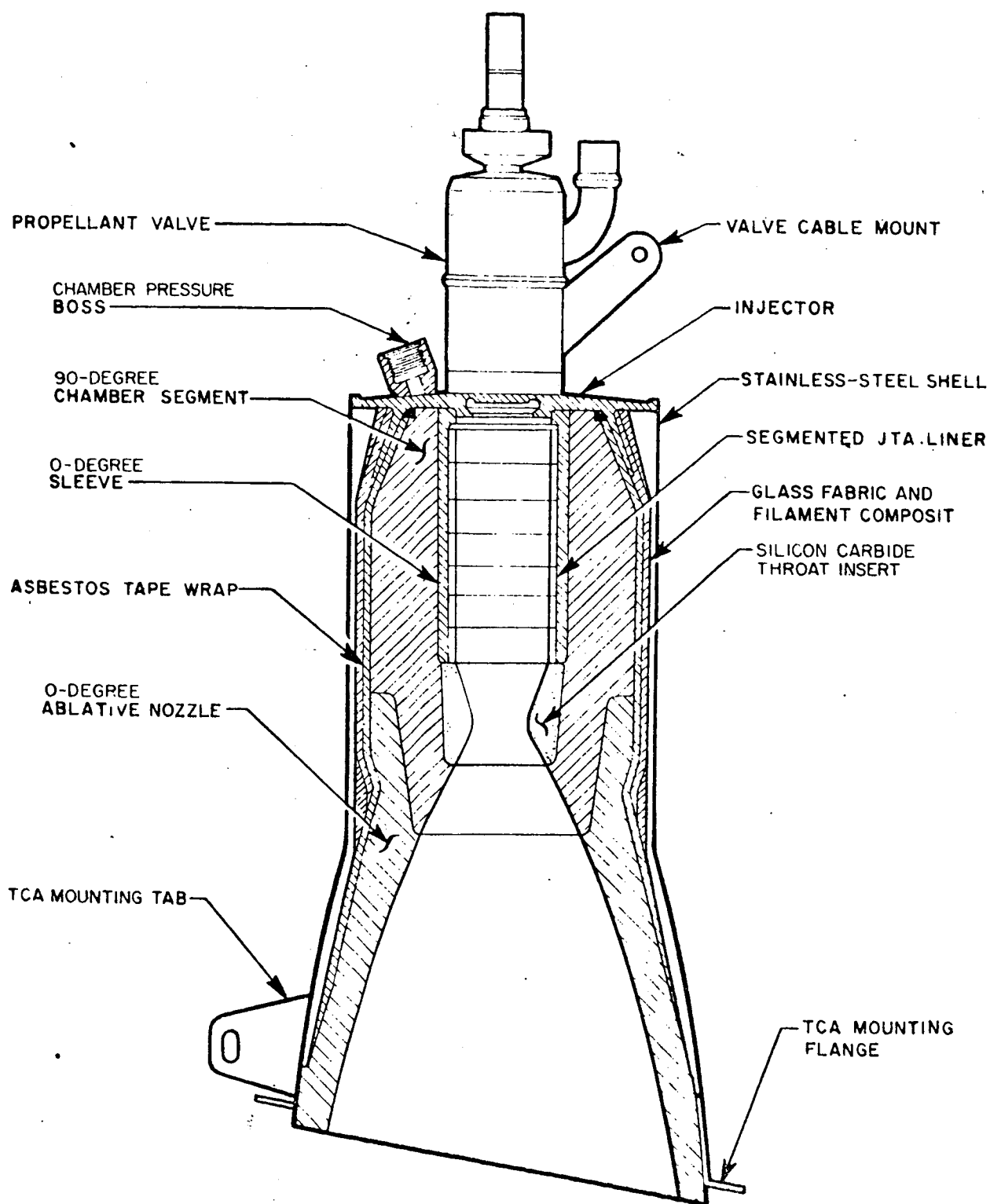
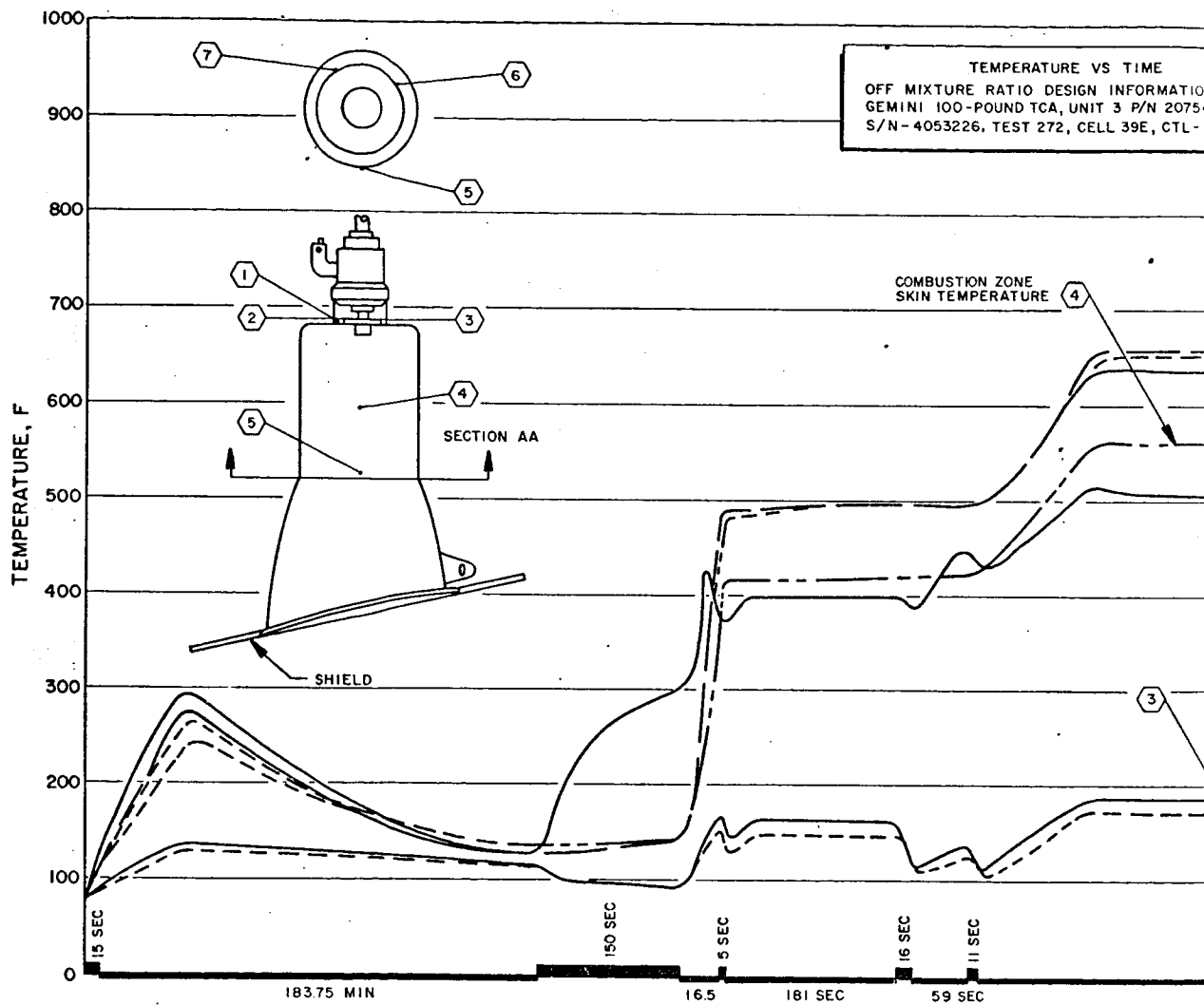


Figure 19. Gemini 100-Pound-Thrust TCA, Cross-Sectional Diagram

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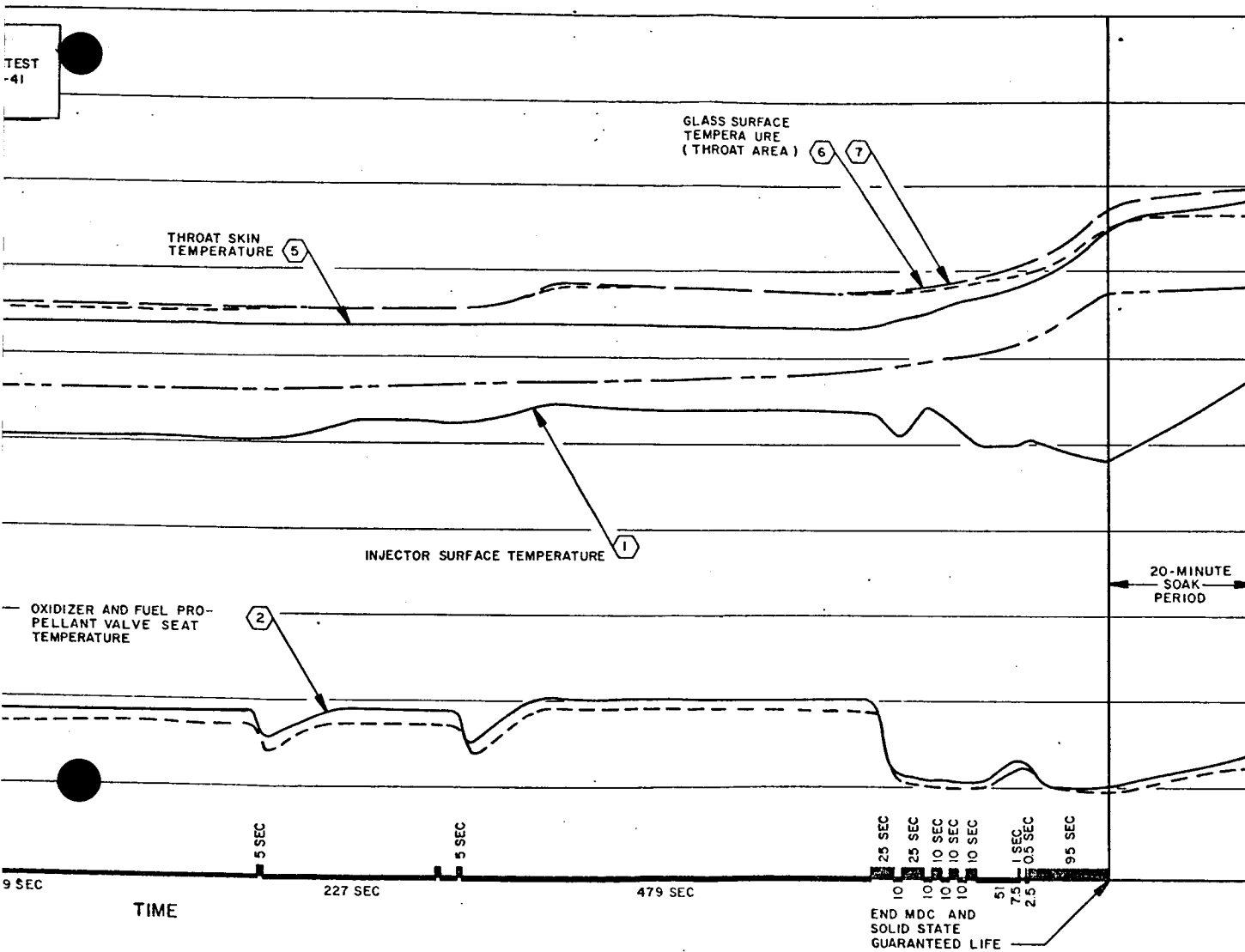


Figure 20. Temperature vs Time, Gemini  
100-Pound-Thrust TCA



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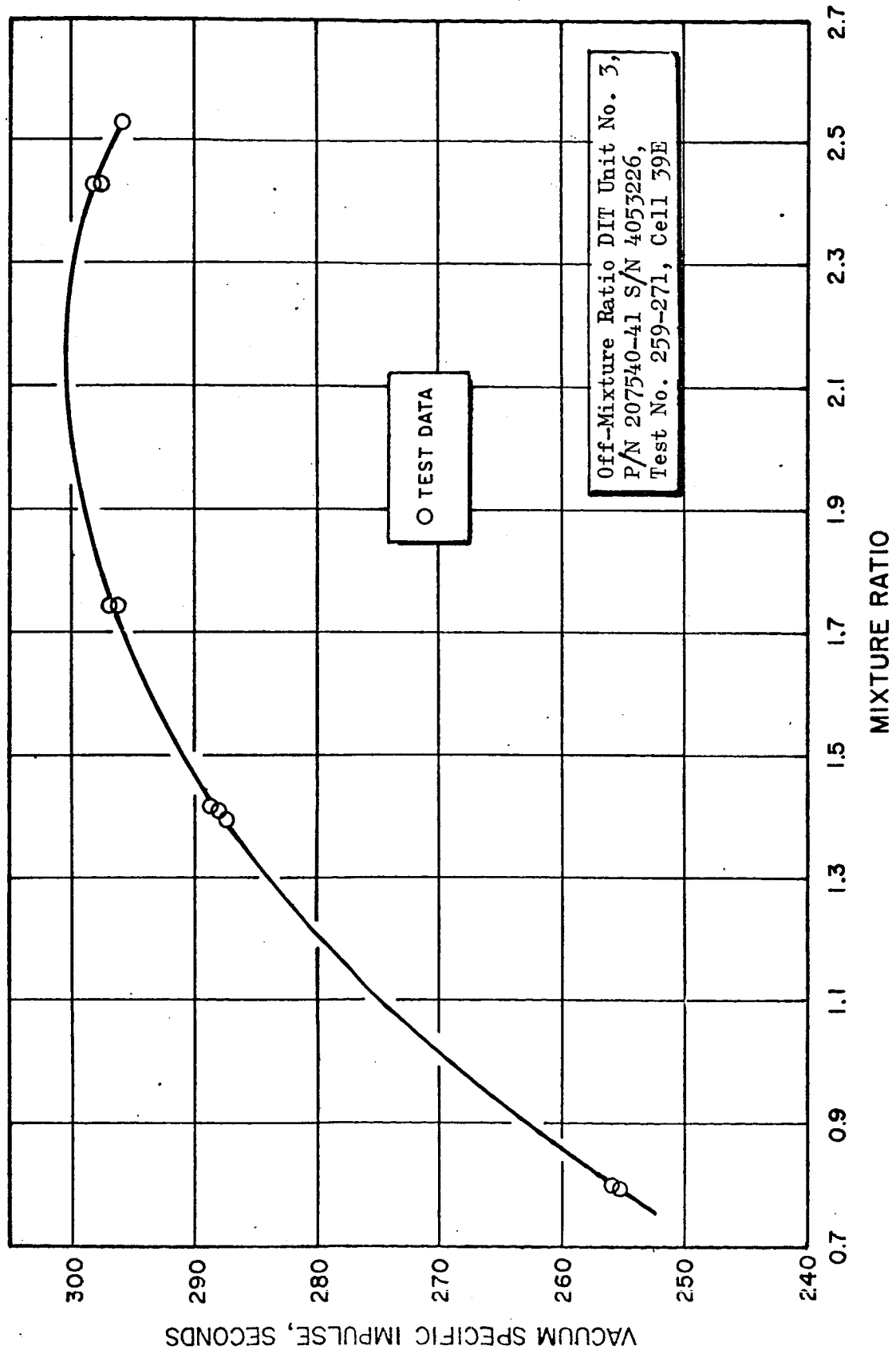


Figure 22.. Vacuum I<sub>s</sub> vs Mixture Ratio

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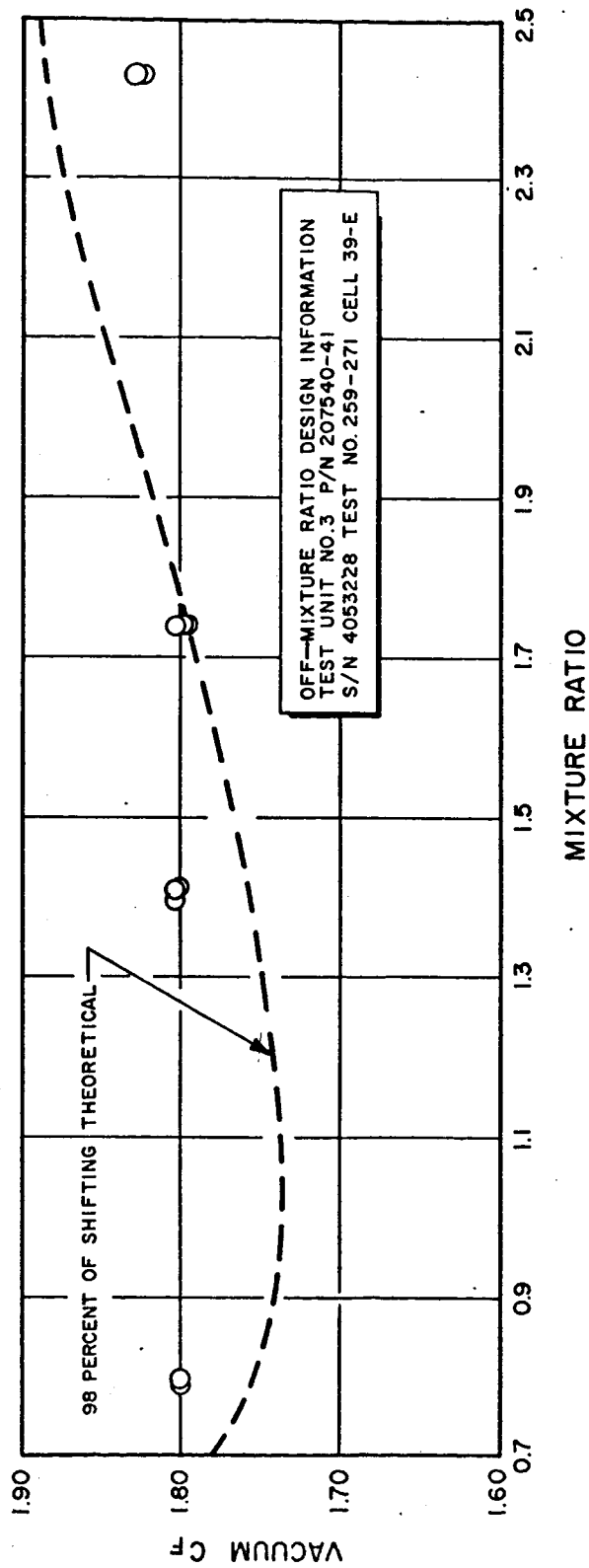


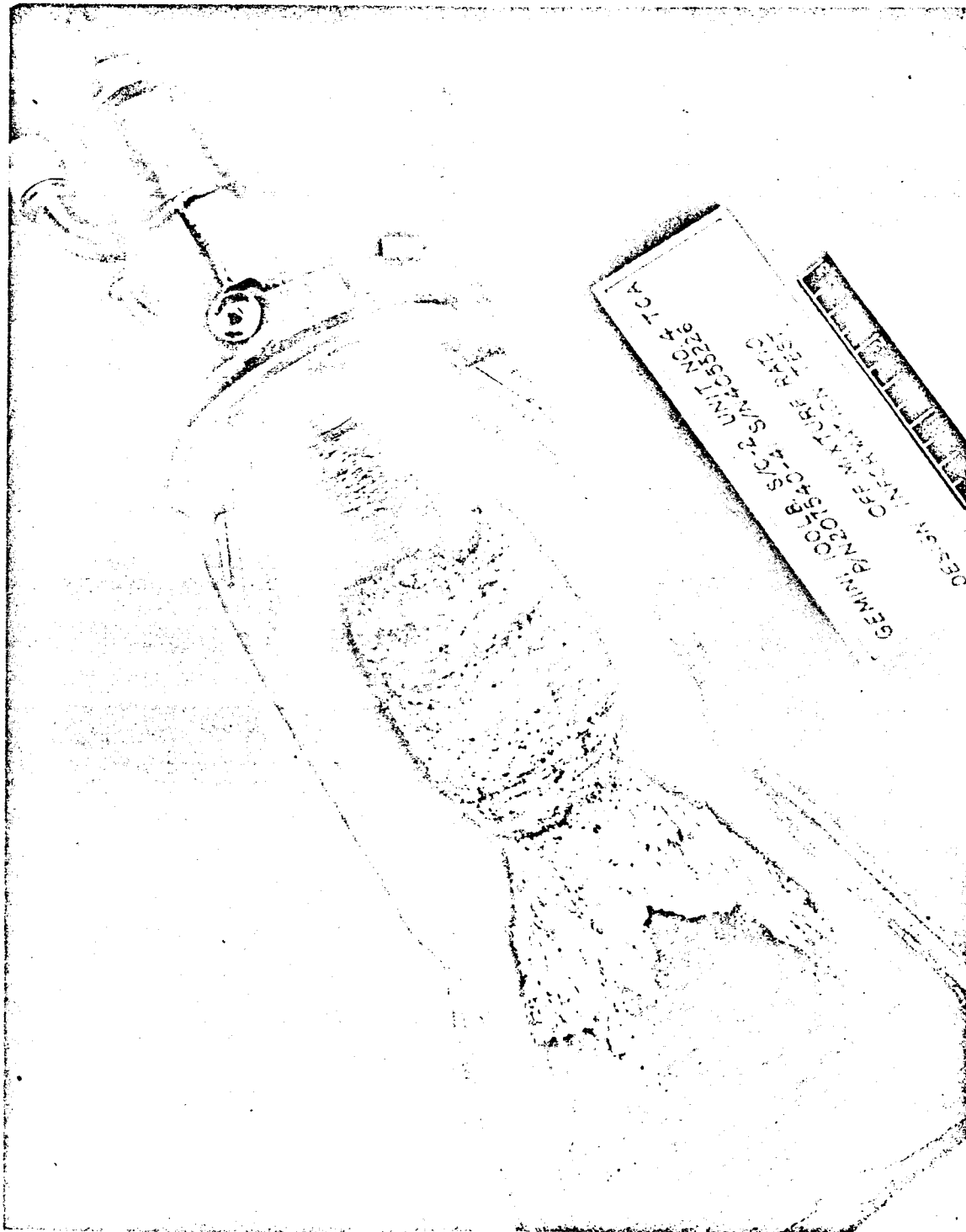
Figure 23. Vacuum  $C_F$  vs Mixture Ratio

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Figure 24. Postfire TCA

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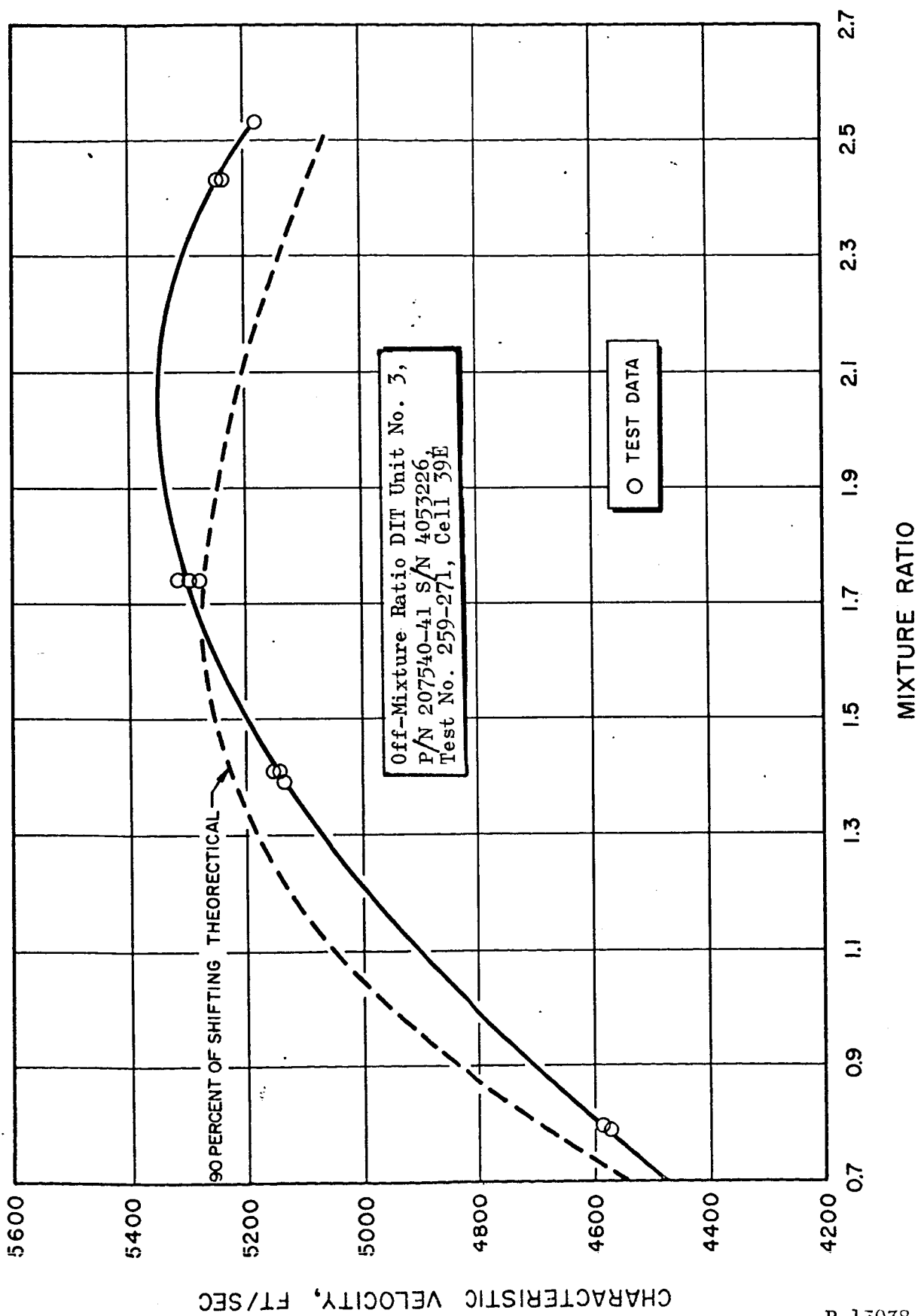
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Figure 21. Characteristic Velocity vs Mixture Ratio

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APPENDIX E

STUCK VALVE TEST

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TITLE: STUCK VALVE DESIGN INFORMATION TEST

INTRODUCTION

A series of stuck valve tests was performed with a Gemini 100 pound flight-weight thrust chamber to determine propellant flow oscillations, chamber pressure oscillations, and the effects on the thrust chamber life. The test consisted of a series of cycles where the oxidizer valve was "open" for  $2.5^{+0}_{-5}$  seconds and "closed" for  $3.5^{+0}_{-5}$  seconds while the fuel valve was held in the full open position. The test procedure was then reversed with the fuel valve cycled while the oxidizer valve remained "open." Testing was accomplished under condition of local ambient temperature and pressure with the thrust chamber in a 17 degree from the horizontal nozzle up position.

SUMMARY

The stuck valve design information test was completed successfully with no oscillations in chamber pressure or propellant flow during the test. The actual engine burntime (both valves open simultaneously) was 65.6 seconds with normal engine performance throughout the cycling of the valves. The posttest inspection showed the thrust chamber assembly to be in excellent condition. X-rays taken of the chamber revealed several delaminations of the ablative body.

DISCUSSION

The test chamber was a Capsule II 100 pound flightweight thrust chamber P/N 207540-41, S/N 4053551, used for "test only" because of a cracked

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liner segment. Testing was conducted on July 8 1964, at CTL III, Cell 28D, at local ambient conditions, and with the thrust chamber in a 17 degree from the horizontal nozzle up position. Two calibration tests at a 1.6 mixture ratio preceeded the stuck valve tests to verify engine performance (Table 1). Sixteen (16) oxidizer valve pulses were conducted on the TCA with the fuel valve held in the full open position. Fuel flow was 0.1257 lb./sec. during the time that the oxidizer valve was open and increased to 0.1685 when the oxidizer valve was closed. Fuel expended was to be  $9^{+2}_{-0}$  lbs. requiring a total "open" time for the fuel valve of  $67 \pm 5$  seconds. The tests actually exceeded the requirements as the total fuel expended was 11.8 lbs and the total "open" time for the fuel valve was 79.25 seconds. The pulsing of the valves was done manually and the additional time of test was due to human error. The data is tabulated for test (1149) in Table 1 and graphically represented in Figure 25. The engine was then inspected and found to be in excellent condition with no change in the size of the crack in the liner segment. Test Number 1150 was the second portion of the stuck valve test and was run after a cool-down period of one hour. The fuel valve was cycled twelve (12) times while the oxidizer valve was held in the full "open" position. Oxidizer flow was 0.2052 lb./sec. while the fuel valve was open and increased to 0.2583 when the fuel valve was closed. The test required  $12^{+2}_{-0}$  lbs. of oxidizer to be expended and was run for 58.37 seconds deplting 13.44 lbs. of NT0. All data is tabulated in Table 1 and graphically represented in Figure 26. The increases in flowrates, during the time when one valve was open and the other closed, were due to the differential pressure drop caused by the absence of chamber pressure.

#### PERFORMANCE

The performance of the thrust chamber was satisfactory in all respects. Valve response times were consistant and only normal oscillations of chamber pressure and propellant flow were encountered during the tests.

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C\* efficiency (Figure 27) was two percent higher for the stuck valve tests than for the calibration tests. This was attributed to the amount of propellant puddling in the combustion chamber between pulses as a result of the nozzle up position. Total accumulated burntime (actual firing time) was 65.6 seconds.

#### POST-TEST CONDITION

No added damage was apparent to the TCA liner segments at the completion of the tests. The throat insert and the explosion proof injector were found to be in excellent condition. X-rays show several delaminations of the 90 degree ablative body (Figure 28) from the third liner segment to the trailing edge of the throat insert. A portion of the ablative, 0.140 inches wide, downstream of the throat insert at the bond lines, is missing, but is a normal occurrence for this type of thrust chamber.

#### CONCLUSIONS

The results of the Stuck Valves tests show that valve oscillations or lag at the start or shut-down of firing by either valve will not cause malfunctions of the thrust chamber. Performance will be such that the engine will perform within specification as to efficiency and life. Structural integrity of the ablative is such that the TCA will continue to perform within specification life even though it contains many delaminations.

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~~CONFIDENTIAL~~TABLE 1PERFORMANCE DATA

Test No.	P <sub>c</sub> psia	W <sub>o</sub> lb/sec	W <sub>f</sub> lb/sec	M/R o/f	C* ft/sec	ηC* %	Duration, sec.
1147	137.4	0.2076	0.1266	1.64	5234	89.2	3.2
1148	136.4	0.2032	0.1276	1.592	5243	89.5	3.2
1149	139.4	0.2052	0.1257	1.632	5363	91.6	79.25
Fuel Valve Open Oxidizer Valve Pulsed							
1150	139.8	0.2052	0.1273	1.612	5353	91.4	58.37
Oxidizer Valve Open Fuel Valve Pulsed							

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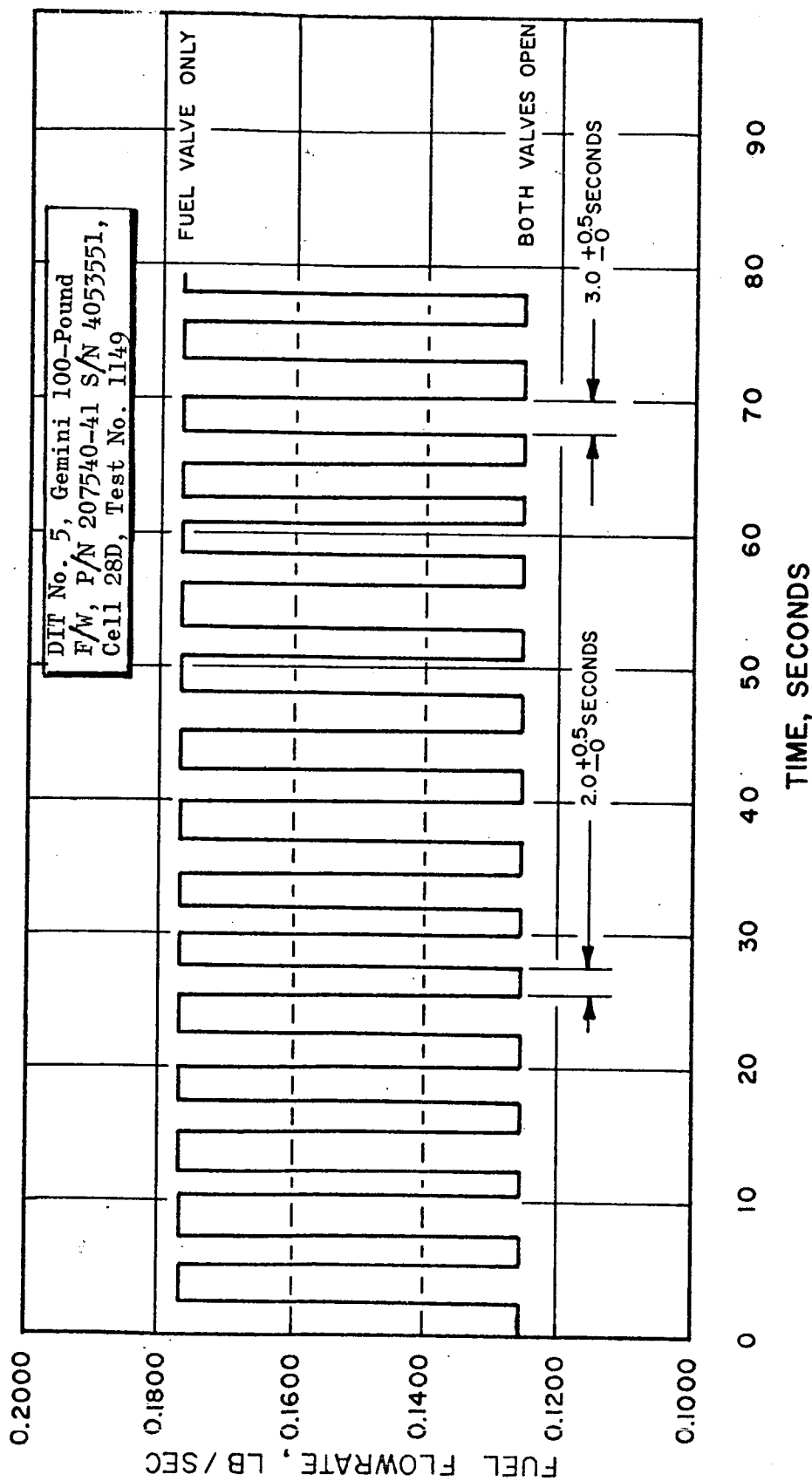


Figure 25. Fuel Flowrate vs Time; Fuel Valve Stuck Open, Oxidizer Valve Pulsed

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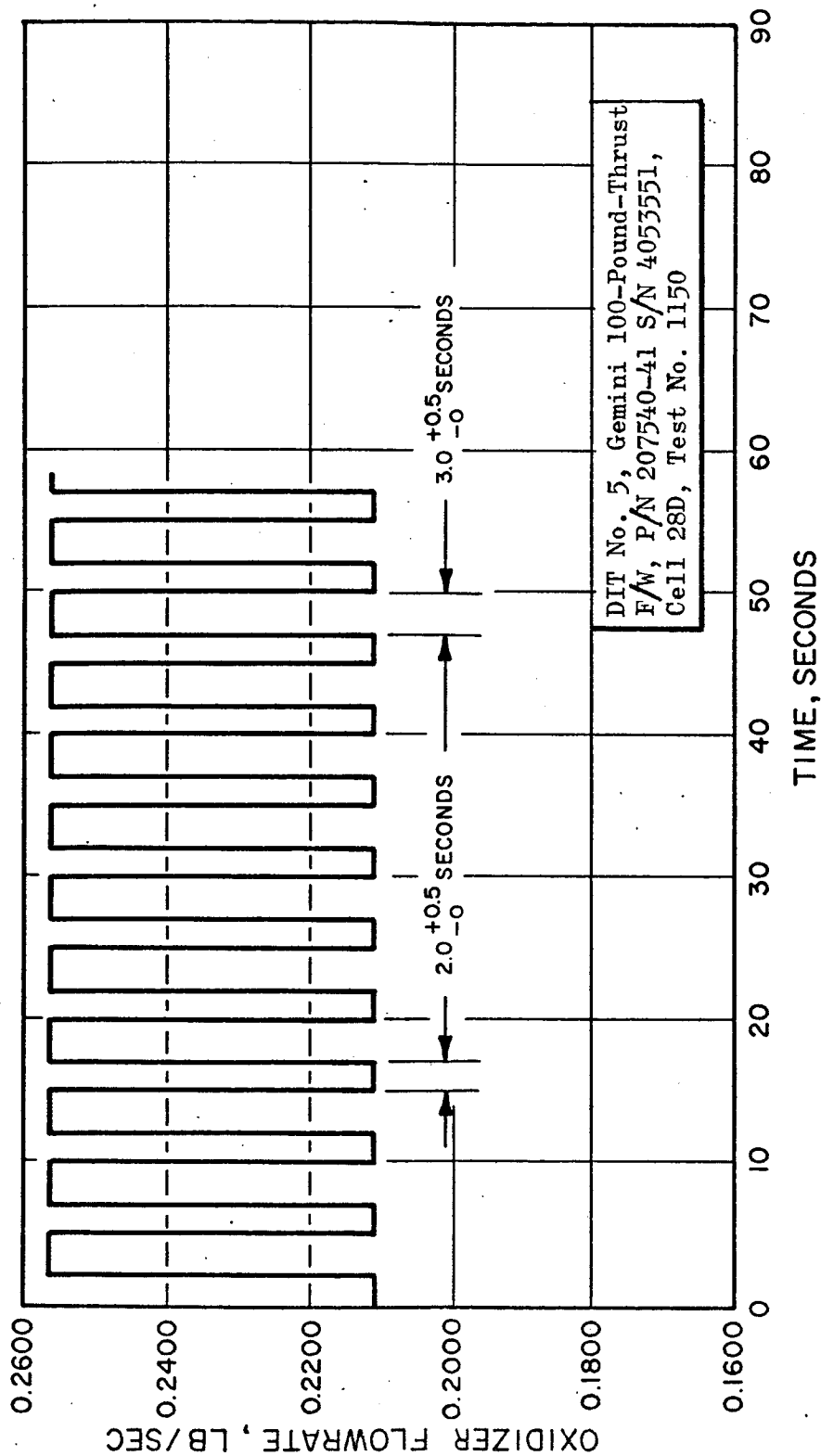


Figure 26. Oxidizer Flowrate vs Time, Oxidizer Valve Stuck Open,  
Fuel Valve Pulsed



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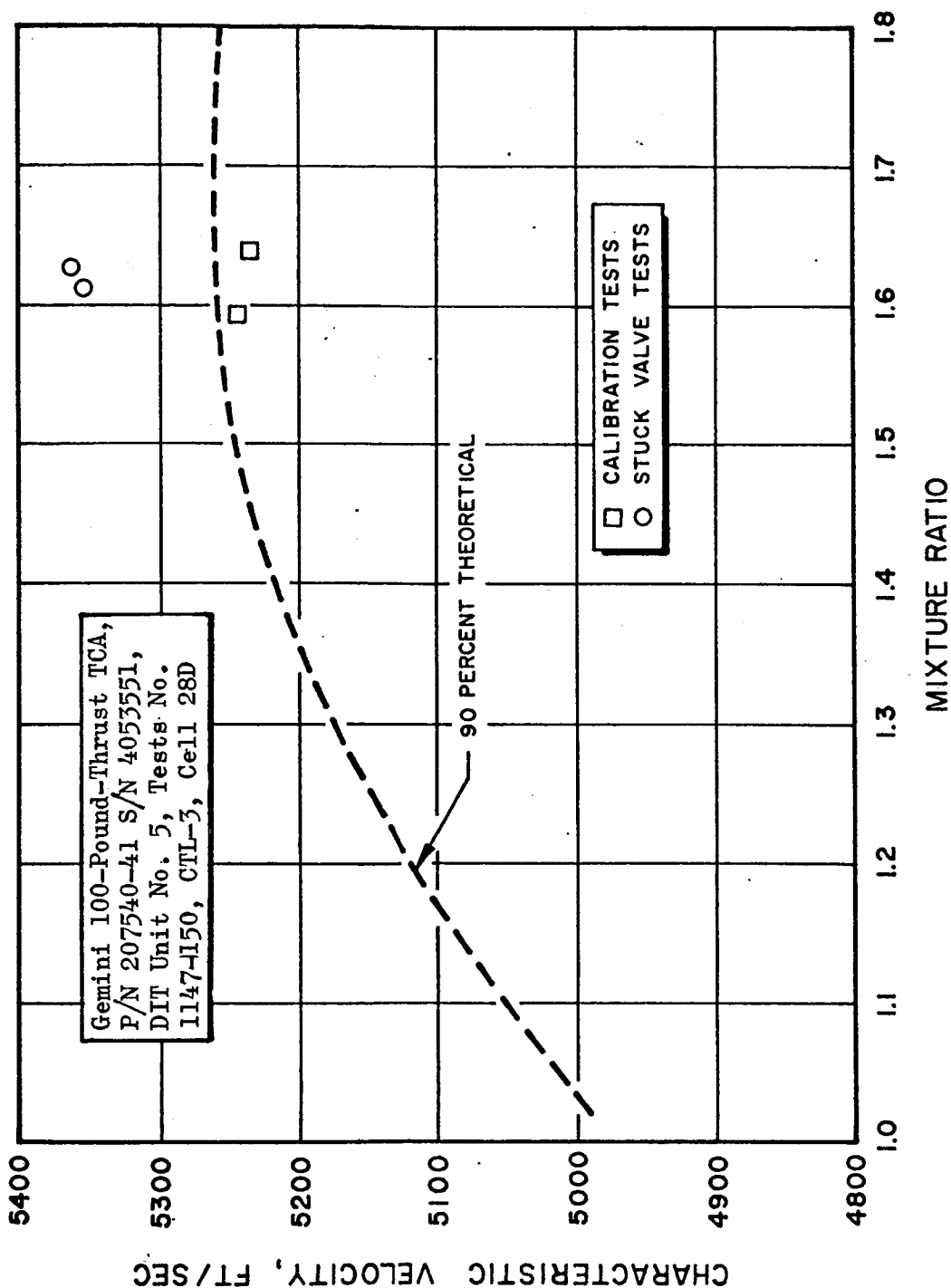


Figure 27. Characteristic Velocity vs Mixture Ratio,  
Stuck Valve Test

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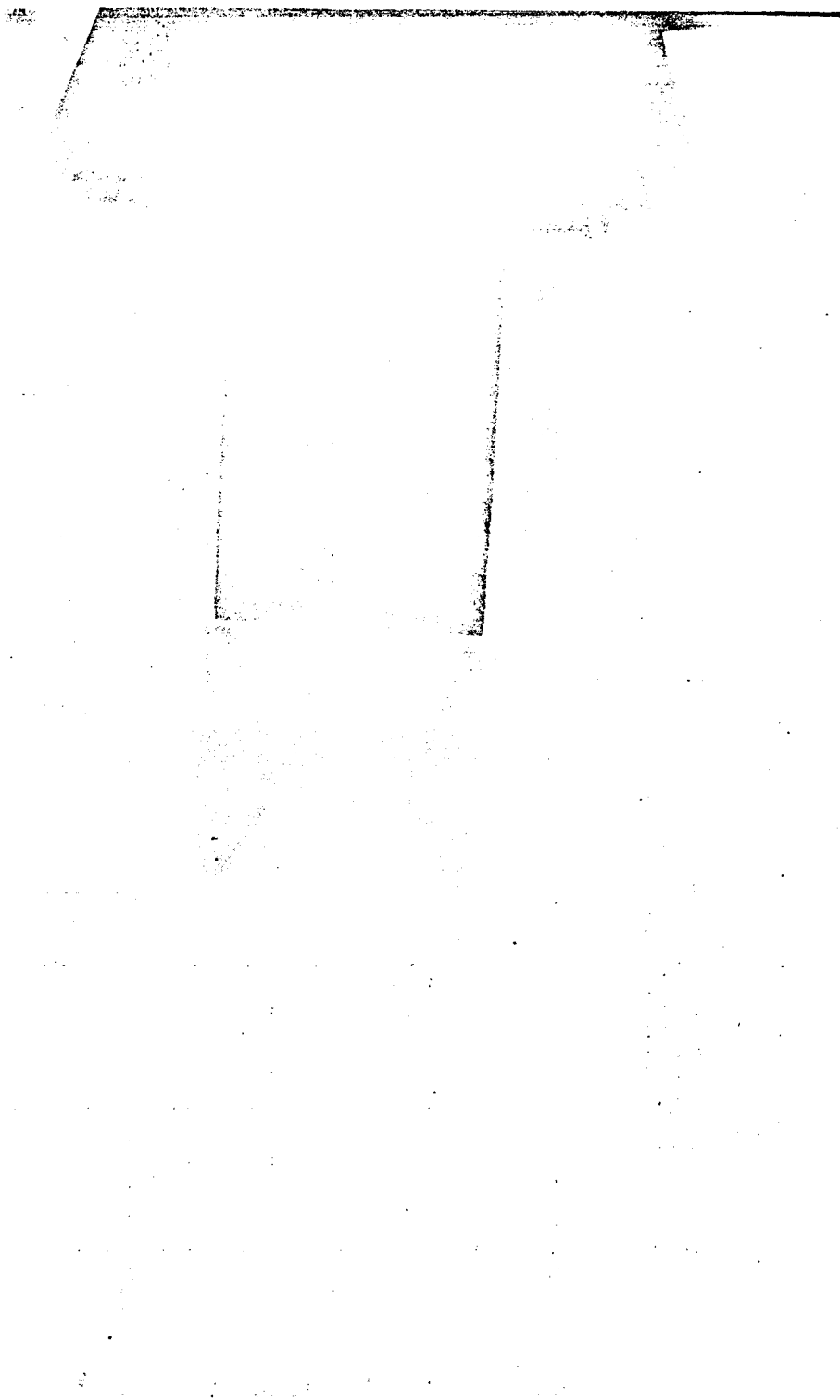


Figure 28. X-Ray Showing Delaminations

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APPENDIX F

HIGH-TEMPERATURE TEST

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~~CONFIDENTIAL~~TITLE: HIGH TEMPERATURE TEST - DESIGN INFORMATION TEST

- Reference: (a) SEM 4390-4017, "Gemini 100 lb Injector Fuel Manifold Distribution Problem", dated 19 May 1964.

INTRODUCTION

On 14 August 1964, a 100 lb S/C 2 TCA was tested to determine the effects of high environmental and propellant temperature on thrust chamber assembly, valve performance, and operation.

Three (3) steady state firing and a series of ten (10) one-half second pulses were conducted in Cell 39K, CTL-III, SSFL, at simulated pressure altitudes above 100,000 feet. The thrust chamber was subjected to an environmental soak temperature of +200°F. The propellants used for this test series were heated and maintained at +165°F. The thrust chamber had previously completed the stuck valve design information test, which resulted in 65.6 seconds of burn time.

SUMMARY

A Gemini S/C 2 100 lb OAMS thrust chamber assembly was tested after it was subjected to an environmental test temperature of +200°F. During the test the propellant temperatures were maintained between +165 to +175°F. The thrust chamber was calibrated to nominal chamber pressure and mixture ratio.

Both steady state and pulse tests under environmental soak conditions and high propellant temperatures were successfully completed. Overall thrust chamber performance was unaffected as a result of environmental soak and high propellant temperatures. The response characteristics of the propellant valves were normal.

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Post-test analysis inspection showed six of the seven liner segments in the combustion zone were missing and one small eroded place in the inside diameter of the injector splash plate occurred during the test series. Pre-test cracks in the throat did not change as the result of the test. Total accumulated burn time for this TCA was 147.0 seconds, including the previous test series.

## DISCUSSION

### Chamber Configuration

Gemini S/C 2 100 lb flightweight TCA P/N 207540-41, S/N 4053551. The thrust chamber employed a seven (7) segmented JTA refractory liner, 90° chamber segment, zero degree wrap sleeve, a solid SiC throat insert, and a zero degree wrap exit nozzle, as shown in the enclosed Figure 29. In addition, four Iron-Constantan thermocouples were installed on the thrust chamber to measure oxidizer and fuel propellant valve seat temperature, injector surface temperature, and glass wrap surface temperature (throat area).

The general hardware condition after the previously tested stuck valve DIT was good. There were two radial throat cracks and one longitudinal crack in the second liner segment adjacent to the  $P_c$  hole. X-ray analysis showed numerous delaminations of the 90° ablative material but based on the extent of the delaminations the chamber did not appear to be damaged. The basic structural condition of the TCA was quite good, with the exception noted above.

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~~CONFIDENTIAL~~Test Description and Performance

The thrust chamber assembly was mounted in the standard nozzle down position in the test stand. A heater blanket was then wrapped around the thrust chamber excluding the valves, in order to obtain the environmental +200°F soak temperature. In addition, the thrust mount temperature was recorded and the thermocouple was attached in the vicinity of the TCA valves.

The propellants were heated by a "hot water jacketed" accumulated and feed system. The TCA environmental soak temperature was stabilized at +200°F and the propellant temperature was controlled to +170°F prior to firing the first steady state test. All test temperatures were recorded and their locations are graphically plotted and indicated in Figure 30.

The testing sequence was conducted as follows:

- a. Fire steady state for a duration of thirty (30) seconds.
- b. Allow the TCA to soak until the oxidizer valve seat temperature peaks, then fire steady state for a duration of twenty (20) seconds.
- c. Repeat steps in b. once.
- d. Allow the TCA to soak until the oxidizer valve seat temperature peaks, then conduct a series of ten (10) tests of 0.5 second duration "on time" separated by 10 seconds "off time" between each firing.

Both steady state and pulse tests under environmental soak conditions and high propellant temperatures were successfully completed.

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Performance data from the calibration test are as follows: Mixture ratio = 1.61,  $C^* = 5416$  ft/sec (92.5% of shifting theoretical), and  $C_f = 1.7643$  (98.8% of shifting theoretical). The resultant  $I_{sp}$  was 11 seconds above nominal, compared with other Gemini 100 lb thrust chambers'  $I_{sp}$  performance.

#### Post-Test Hardware Condition and Analysis

A cursory visual thrust chamber inspection was made immediately after the hot temperature test. One additional longitudinal crack and a new circumferential crack occurred in the throat insert during this test series, with heavy deposits of the segmented liner oxide just downstream of throat area. Six plus part of the seventh segmented JTA liner were eroded from inside the combustion zone. The erosion in one longitudinal area extended through the zero degree wrap sleeve in line with the fuel inlet port of the injector. This longitudinal erosion is typical of this thrust chamber configuration, however, in this particular test the injector splash plate started to erode in this local area.

Pre and post test measurements of the throat diameter indicated that no dimensional change had occurred.

Analysis of the oscillograph records showed no propellant valve mismatch. The inlet pressure deflections and response characteristics were normal in all respects. The results of pulse testing, again indicated no abnormal transient characteristic. The condition of the thrust chamber combustion zone and injector splash plate, in conjunction with the accumulated burn time for this test series and previous test series conducted, indicate that thrust chamber life condition would not have resulted with nominal propellant temperatures.

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Analysis of what caused the injector splash plate to erode in the local area, between orifice No. 1 and No. 16 shows it is necessary to know the fuel manifold distribution characteristic of the injector (SEM 4390-4017). The injector splash plate erosion is theorized to be a function of the high temperature propellants. The oxidizer and fuel propellant temperatures during thrust chamber operation averaged +170°F, where the injector surface temperature during TCA operation varied from 185° to 645°F. Under these temperature conditions, it is theorized that the magnitude of velocity of the fuel flow through orifice No. 1 and No. 16 was low enough to vaporize. Thus, permitting no coolant on the injector splash plate, in turn developed a hot spot, causing erosion.

The time-temperature plot showed the maximum injector surface temperature to be 645°F just prior to the pulsing mode of TCA operation. This temperature was acquired during a ten (10) minute soak period. It took this duration for the oxidizer valve seat temperature to peak to 210°F.

In the combustion chamber (Figure 31) in the area in line with the fuel valve inlet at the injector splash plate between orifice pair No. 1 and No. 16, it is known that the mixture ratio is greater there than the rest of the local injector area. Analysis of this combustion zone erosion shows that during thrust chamber operation, the temperature is of sufficient magnitude in this area that the segmented liner composition begins to melt and form a viscous layer of molten material, which is readily eroded out and is deposited just downstream of the throat area where the temperature region is lower as indicated in Figure 31.

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~~CONFIDENTIAL~~CONCLUSION

Hot temperature testing, as conducted, had no significant effects on thrust chamber performance or start characteristics. Valve performance and operation were normal in all respects to other Gemini 100 lb thrust chamber not exposed to the environmental soak and high temperature propellants.

The burned injector splash plate should be attributed to the hot temperature propellants.

Erosion in the combustion zone in line with the fuel valve is a typical failure mode with this injector configuration. However, the burned injector splash plate was a contributory factor.

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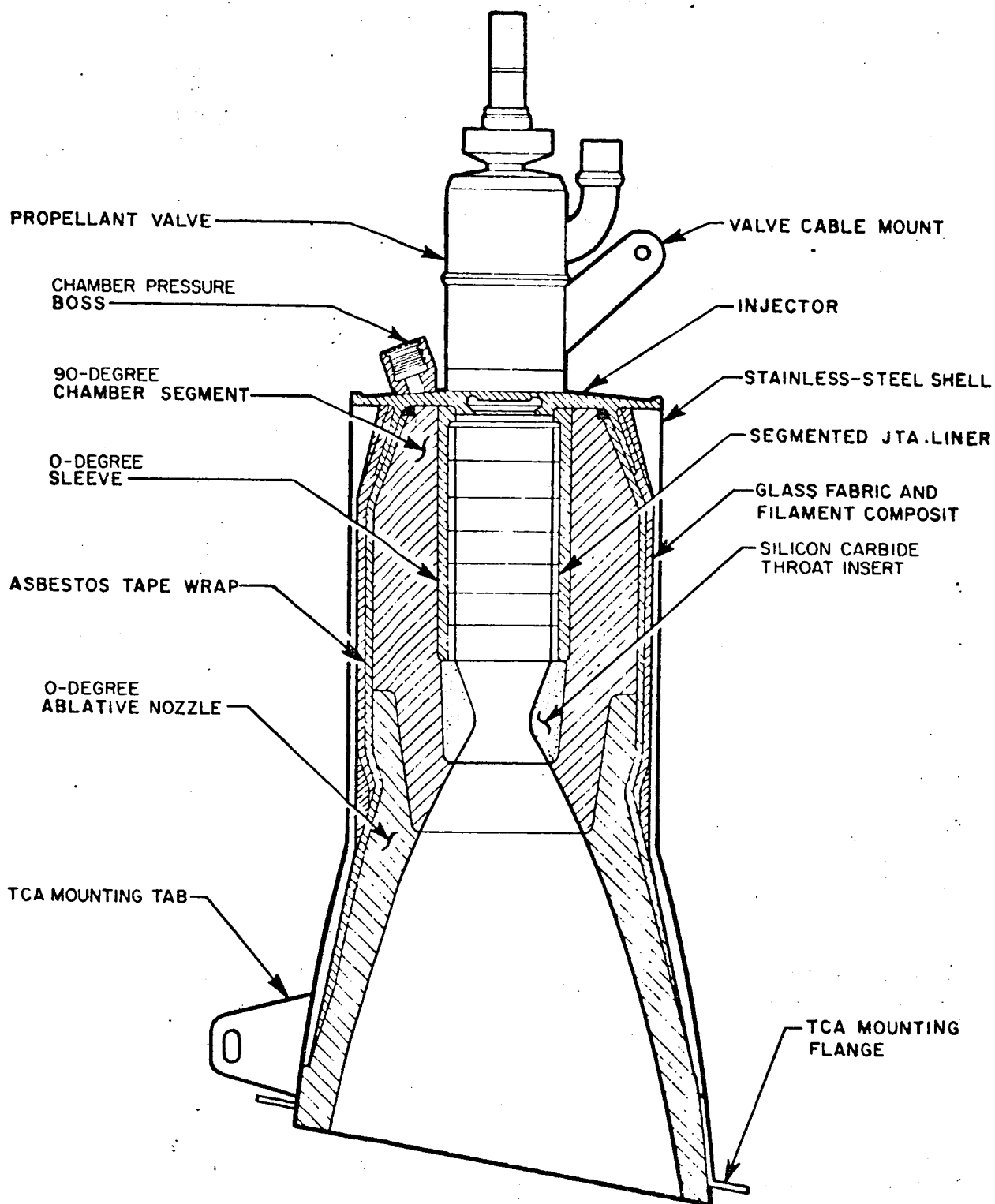


Figure 29. Gemini 100-Pound-Thrust TCA, Cross-Sectional Diagram

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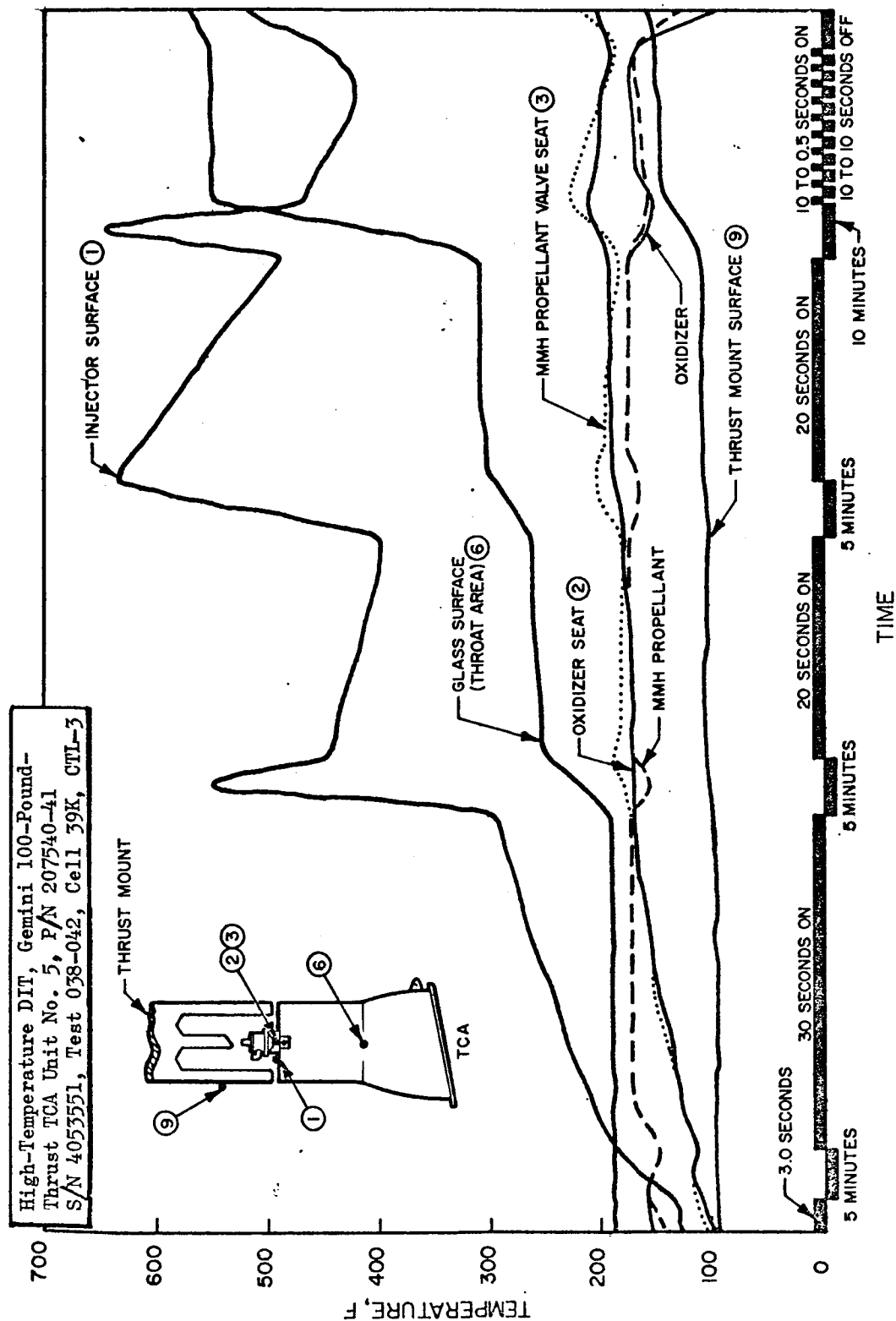


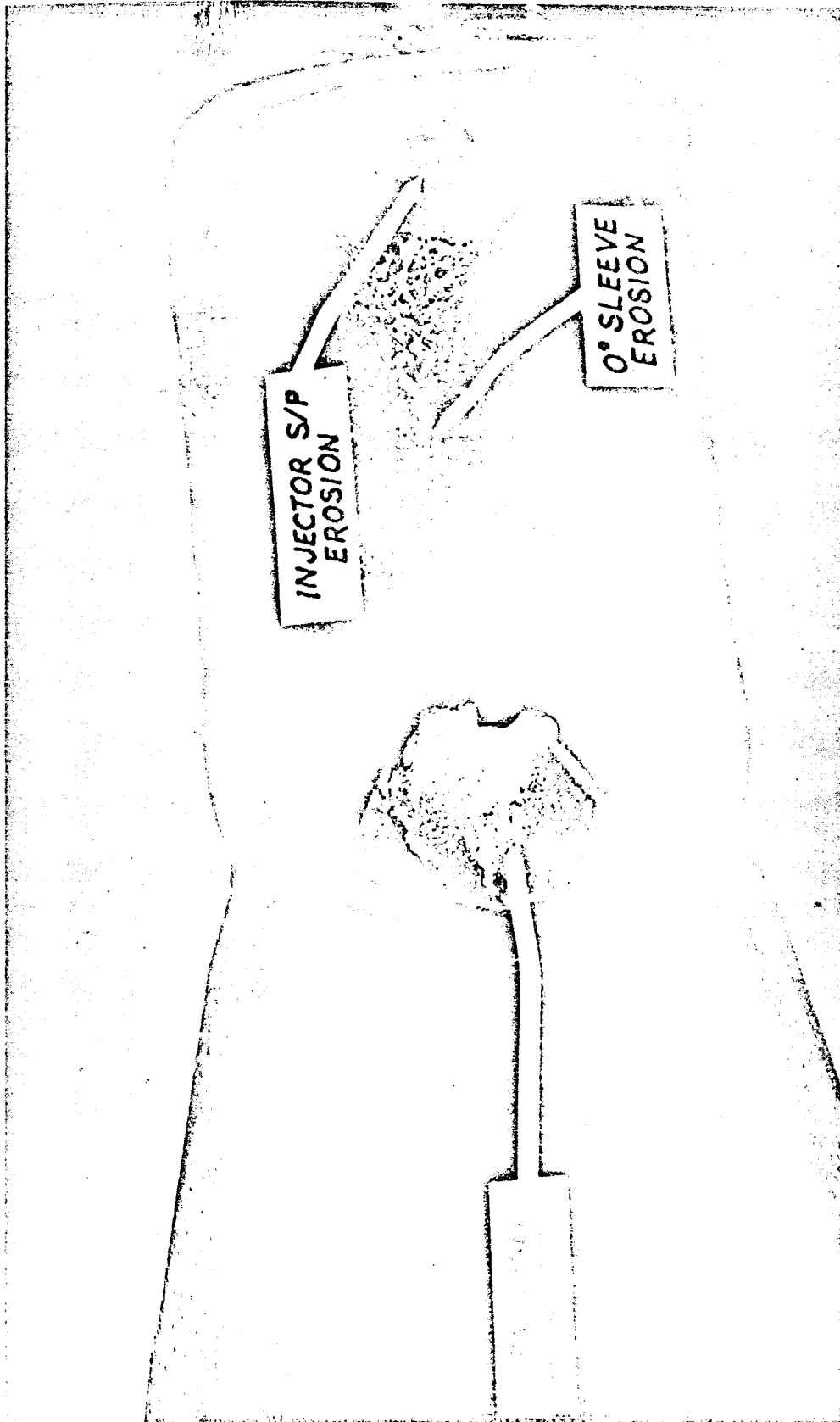
Figure 30. Temperature vs Time

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Figure 31. Combustion Chamber Erosion Areas

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APPENDIX G

PLUGGED INJECTOR TEST

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TITLE: PLUGGED INJECTOR DESIGN INFORMATION TEST

INTRODUCTION

A 100 lb flightweight thrust chamber assembly of the S/C 2 - 5 design was hot-fire tested with four of the oxidizer orifices and four of the fuel orifices welded closed. The orifices that were plugged were picked at random with the only requirement being that they not be impinging holes (oxidizer and fuel matched pairs). Previous to the plugging of the orifices, a series of 3.2 second calibration tests was conducted to establish engine performance and to determine required inlet pressures. After the orifices were plugged, another series of 3.2 second tests was conducted at the same inlet pressures followed by an Aft Firing mission duty cycle (OCM-208) plus steady state test. The objective of the test was to determine the effects of plugged orifices on TCA performance and life.

SUMMARY

The aft firing mission duty cycle plus steady state test, on the 100 lb flightweight TCA with a plugged injector, was terminated after 47.16 seconds of the planned 95.5 second steady state portion at the end of the test. Termination was due to a burnthrough of the ablative chamber and metal shell at the throat station in line with the plugged orifices. The burnthrough overlapped the area of both the plugged oxidizer holes and the area of the plugged fuel holes. The burnthrough was actually a burst through as the temperature had only reached 600°F on the metal shell when the failure suddenly occurred. Total firing time of the chamber at termination was 370.86 seconds. Post-test inspection after sectioning showed a large portion of the ablative combustion chamber wall burned away as were the last liner segment and part of another. About half of the throat insert had been "washed" away and a large amount of

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silica buildup was evident in the throat. Analysis shows the burnthrough to be a direct cause of the extreme amount of oxidizer present in the area of burnthrough.

#### DISCUSSION

The thrust chamber used for the plugged injector DIT test was a Capsule 2 - 5 100 lb flightweight TCA, P/N 207540-41, S/N 4056006. Prior to the orifice plugging, a series of 3.2 second calibration tests was conducted on 7 August 1964, at SSFL, CTL-III, Cell 39E. These tests, over a mixture ratio band of 1.34 to 1.69, were accomplished to establish engine performance and inlet pressures. The TCA was then reworked to plug four oxidizer and four fuel orifices. The orifices plugged were four oxidizers in succession and four fuels in succession, but none were matching pairs (Figure 32). The situation simulated by this type of plugging is considered to be the worst possible. An area of fuel rich (oxidizer orifices plugged) and an area of oxidizer rich (fuel orifices plugged) are directly adjacent to each other. The TCA because of the rework was reidentified as P/N X-E.0.R 916463, S/N 4056006.

The plugged injector DIT testing was conducted at SSFL, CTL-III, Cell 39K, on 25 August 1964. First a series of 3.2 second calibration tests, at the same inlet pressures as the tests before plugging of the orifices, was accomplished to determine the difference in engine performance. Following these tests was an OCM-208 aft firing mission duty cycle and steady state test to determine the life capabilities for a TCA with a plugged injector. The test was performed at a mixture ratio of 1.69.

Performance of the TCA prior to the orifice plugging was established at  $C^* = 5221$  ft/sec, 89.2% of shifting theoretical, mixture ratio of 1.61, and an  $I_{sp}$  eight seconds above the nominal (test 290, Table 2). After

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plugging the orifices, test 050 (Table 2) at the same inlet pressures as test 290, showed engine performance to be  $C^* = 4489$  ft/sec, 76.7% of shifting theoretical, a mixture ratio of 1.70, and an  $I_{sp}$  33 seconds below the nominal. The plugging of the orifices decreased the engine performance by 12.5%,  $I_{sp}$  by 14%, and increased overall mixture ratio by 5.6% (Figure 33). Propellant flow increases were evident during the plugged injector tests (Figure 34). All pressure transients and valve responses were identical for the calibration tests prior to and following the orifice plugging.

The test was terminated after 47.16 seconds of the planned 95.5 second steady state test because of a sudden rise in temperature of the glass wrap and metal shell (Figure 35). Total burn time on the TCA was 370.86 seconds. The criteria of the test was to terminate on temperature rather than upon loss of  $P_c$  or thrust. Because of the extremely exaggerated plugged condition of the injector, the TCA was expected to suffer a catastrophic failure such as a burnthrough of the ablative.  $P_c$  and thrust measurements were very erratic during the steady state test, indicating an alternating buildup of material in the throat and then the washing away of the burned up ablative. A complete burnthrough of the TCA did occur after 47.16 seconds of the steady state test.

Post-test x-rays and sectioning of the TCA showed the chamber to be completely burned out in the area of the throat station and almost evenly spaced between the oxidizer rich and fuel rich areas. The total burnthrough encompassed approximately 80 degrees radially and was located from approximately 3:00 to 6:00, using  $P_c$  location as a 12:00 reference (Figure 32). The last liner and a large portion of the throat insert was washed out through the nozzle. The combustion zone was increased substantially and the ablative material was also expelled through

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the nozzle. A large amount of material buildup is evident in the throat section (Figure 36). Also noticeable are several heavy grooves in the liners near the head end of the combustion chamber in line with the plugged fuel orifices. These grooves were caused by the heavy concentration of oxidizer.

A post-test study of the injector flow pattern was accomplished at the medium water flow laboratory. Photos taken show a large concentration (globules) of oxidizer being directed toward the area of burnthrough. The fuel pattern is also off-center and photos of the combined oxidizer and fuel flow show the flow pattern to be heavily concentrated in the area of the burnthrough (Figures 37 and 38).

The increase in oxidizer flow evident during these tests was located in one area where little or no fuel was available. The head end liners in this area were grooved from this concentration of flow and are directly upstream of the large burnthrough. The burnout area actually overlapped from the fuel starved (oxidizer rich) area to the oxidizer starved (fuel rich) area. The combustion zone was increased continuously as the ablative material was burned and expelled through the throat. The combustion and burnout of the ablative continued until the outside wall of the ablative was weak enough to burst. This fact is evident from the temperature recorded on the metal shell directly at the point of burnout. The temperature (Figure 35) was steadily rising to 600°F and then suddenly pegged at 1000°F. Also the condition of the metal shell showed this to be a sudden burst rather than a gradual burnthrough.

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### CONCLUSIONS

The completion of the 100 lb plugged injector DIT confirmed all suspicions as to effects on TCA performance and life. A severely plugged injector will definitely decrease the performance of the TCA by a large amount and will decrease life. Degradation of performance can be correlated with the number of orifices plugged; the more orifices plugged, the lower the performance. Life, however, is not solely dependent upon the number of orifices plugged but also on the dispersion. The severity of the plugged orifices is also a factor to be considered as to life of the TCA. Partially plugged or intermittantly plugged (due to chips in the manifold) orifices should not affect life any great amount but completely plugged orifices, as in this case, will. A few completely plugged orifices is more detrimental to life than several partially plugged orifices. From the analysis of the above test, failure can be attributed to the plugged orifices which allowed the TCA to have a high concentration of both oxidizer and fuel washing the walls.

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TABLE 2PLUGGED INJECTOR DIT NO. 4

100 lb TCA P/N X-E.0.R 916463, S/N 4056006

Prior to Orifice Plugging Cell 39E 7 August 1964						
Test No.	P <sub>c</sub> , psia	M/R, o/f	C*, ft/sec	$\eta$ C*, %	PI0, psig	PIF, psig
289	135.4	1.61	5198	88.9	266.5	252.5
290	148.0	1.61	5221	89.2	288.5	274.5
291	148.4	1.69	5219	89.1	294	267.5
292	146.0	1.34	5077	87.9	269	309.5
293	149.2	1.49	5162	88.5	284.5	295.0
Orifices Plugged Cell 39K 25 August 1964						
047	130.6	1.65	4504	77.0	279.4	276
048	133.0	1.54	4475	76.6	280.4	301
049	134.2	1.74	4484	76.7	300.4	280
050	132.0	1.70	4489	76.7	288.4	275.5
051	131.0	1.47	4446	76.4	269.4	308.5
052	132.0	1.77	4510	77.2	293.1	266.5
053	133.8	1.69	4612	78.8	285.4	273.5



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Chamber Pressure Tap Location

12:00

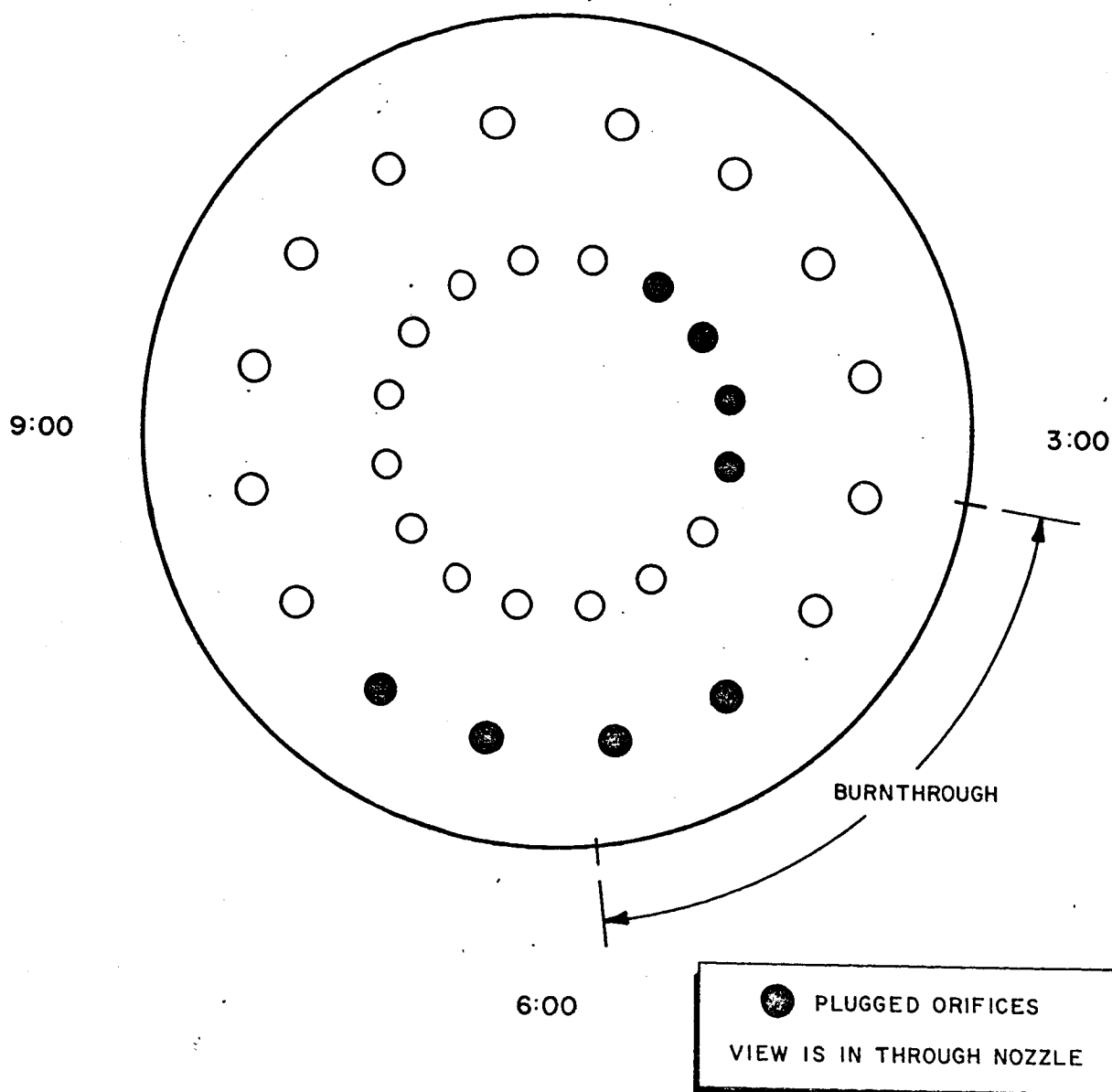


Figure 32. Plugged Orifice Locations

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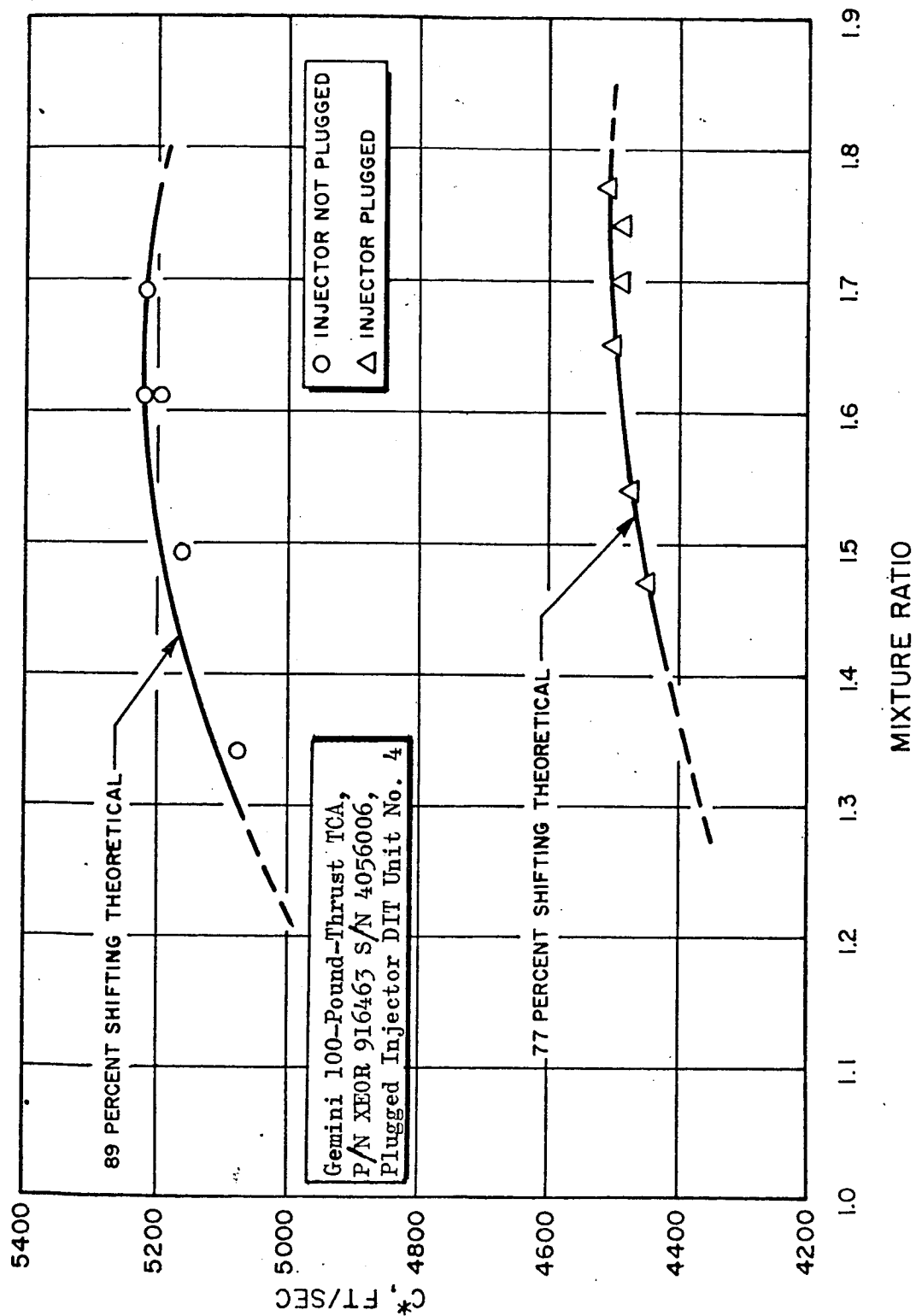


Figure 33. Characteristic Velocity vs Mixture Ratio

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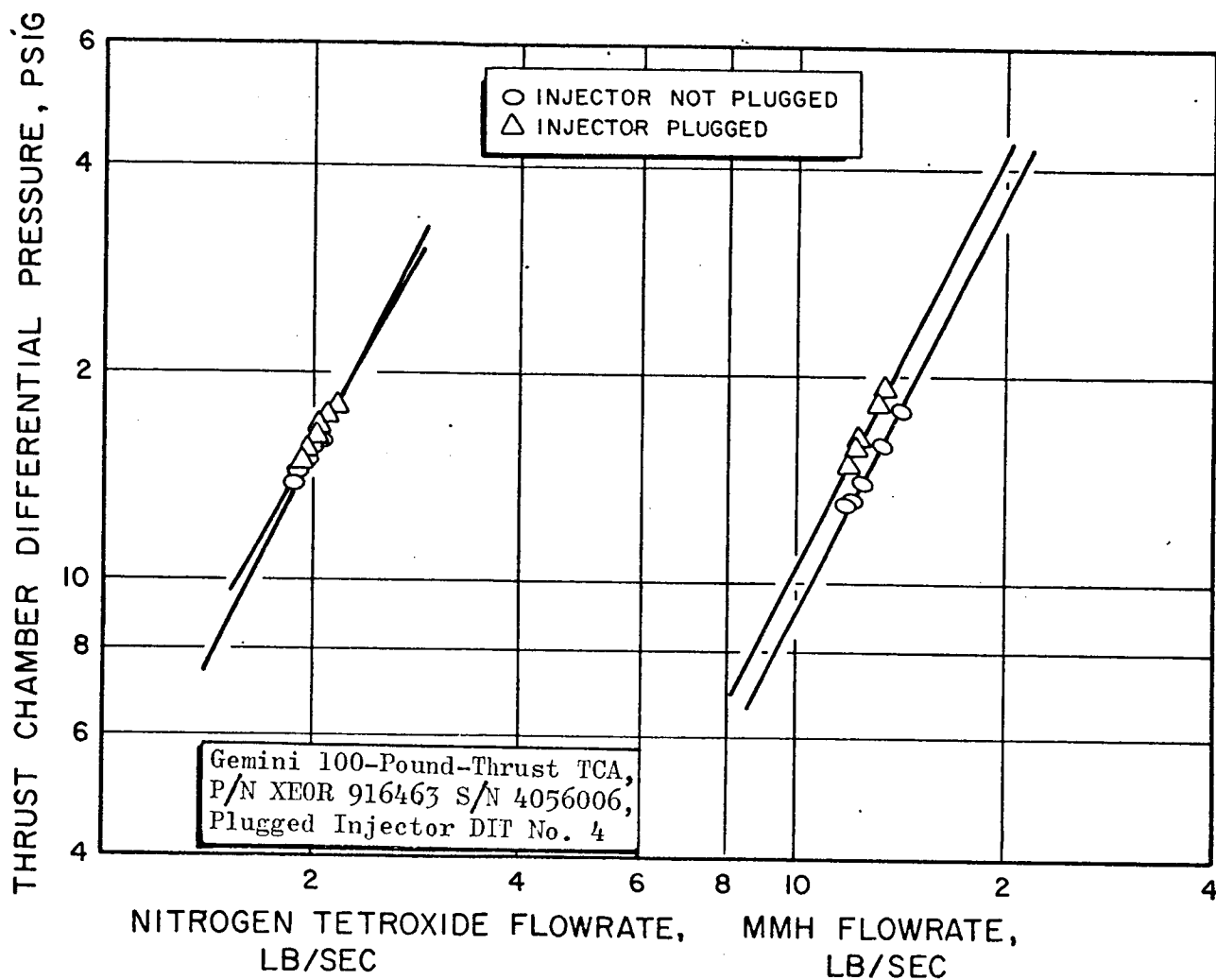
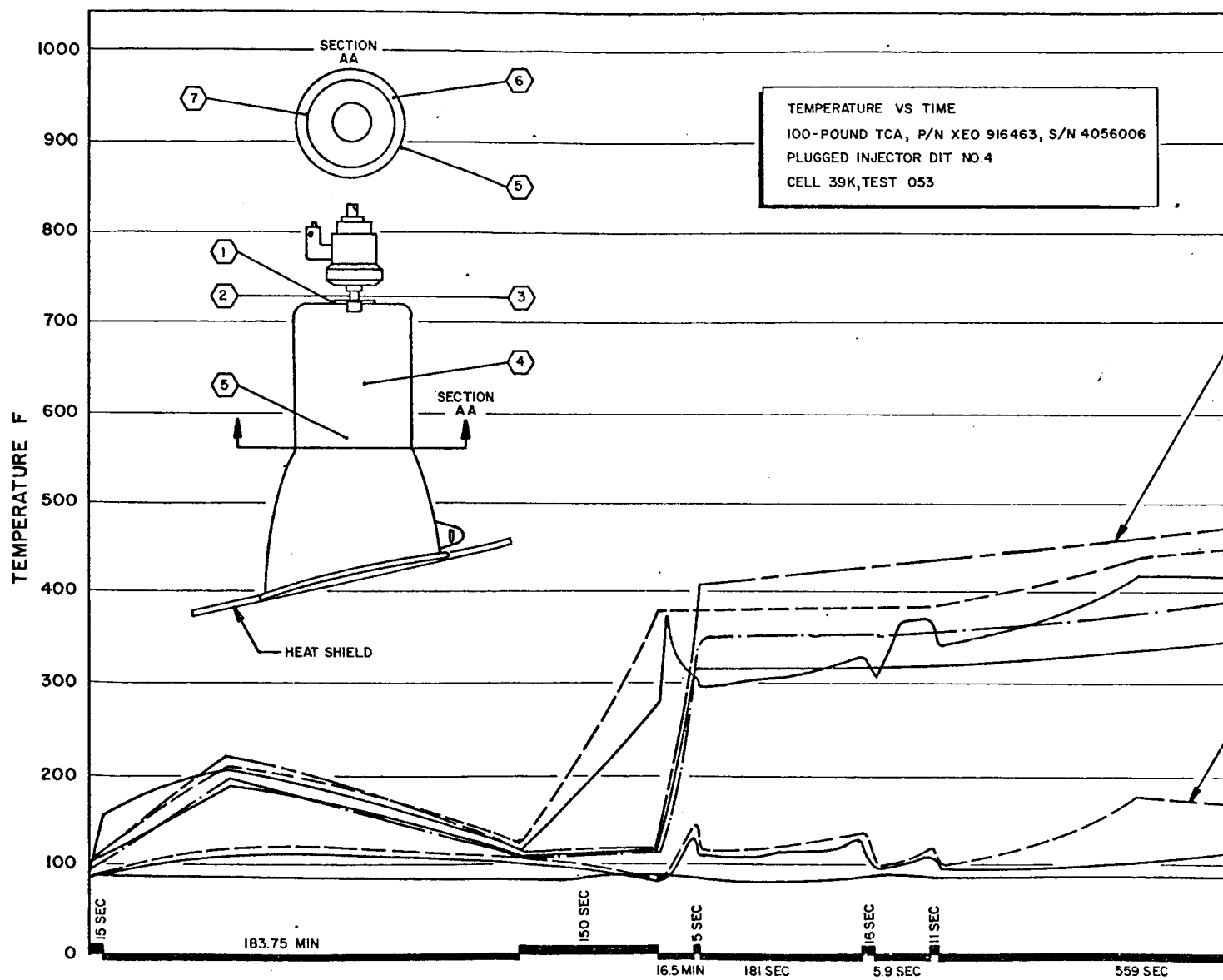


Figure 34. Thrust Chamber  $\Delta P$  vs Flows

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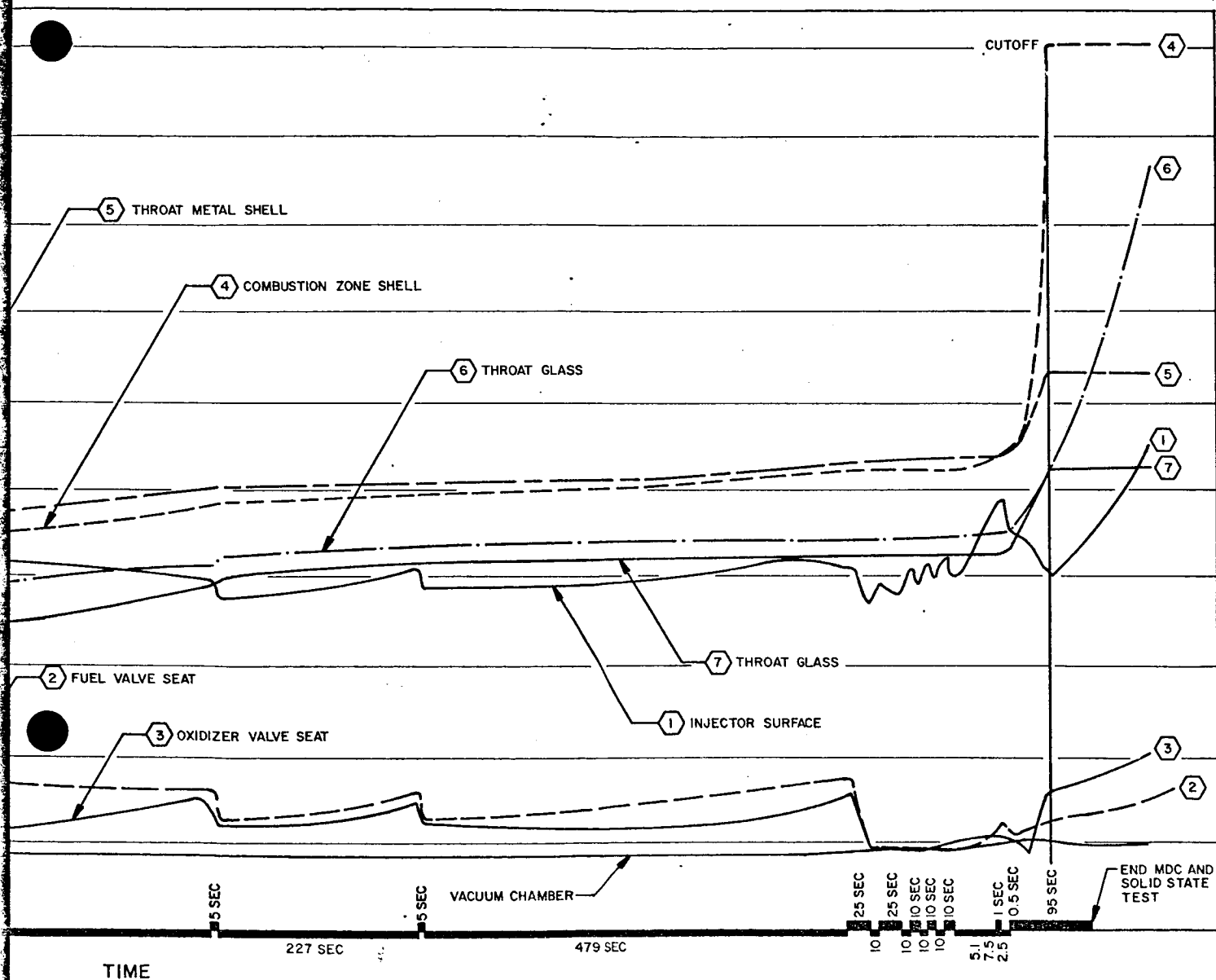
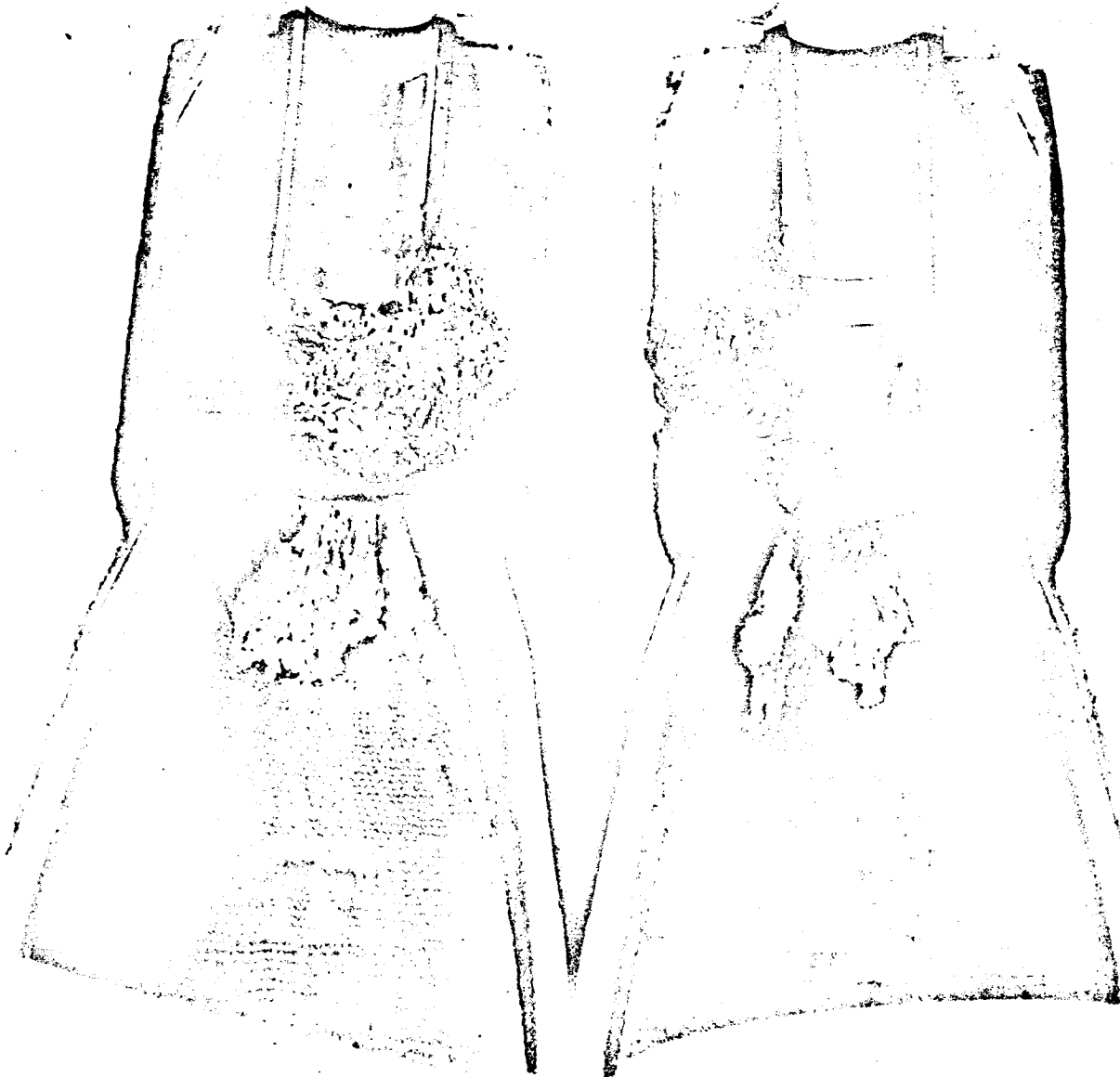


Figure 35. Temperature vs Time

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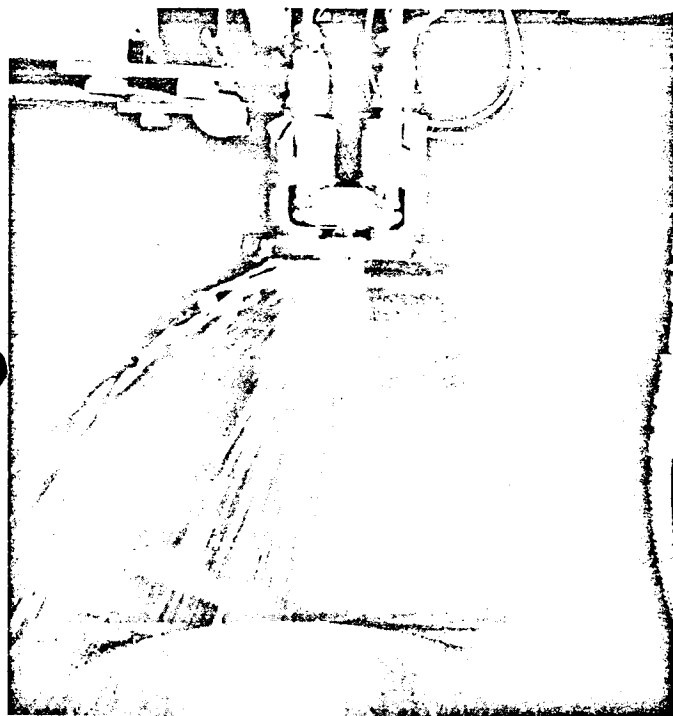
Figure 36. Material Buildup in Throat Section

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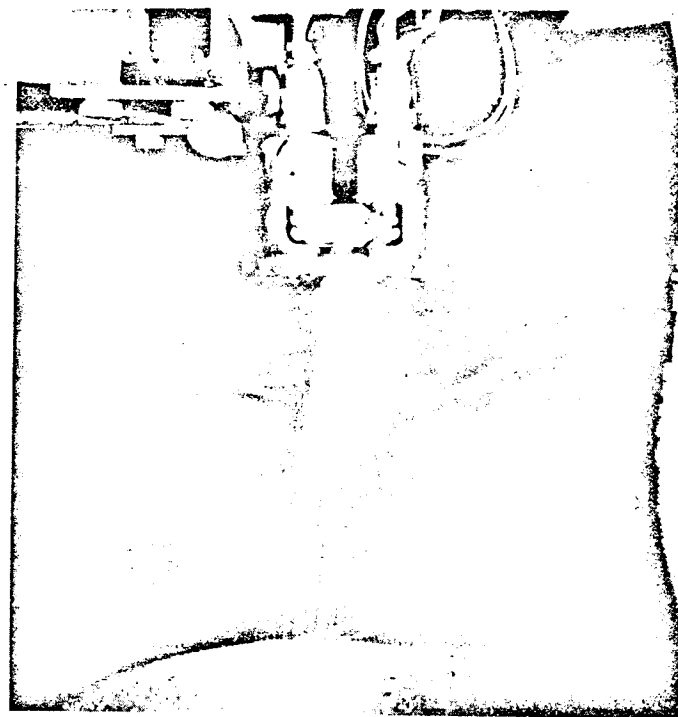
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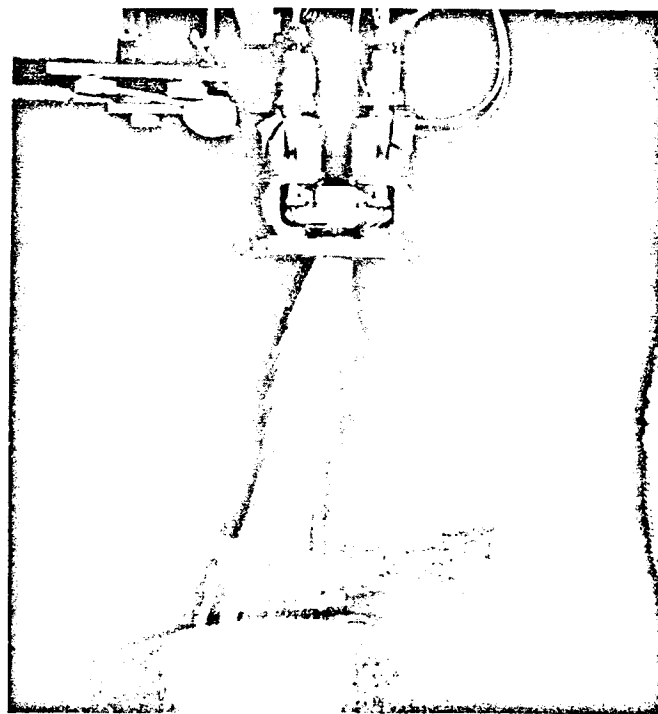
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Oxidizer only



Fuel only



Oxidizer and fuel

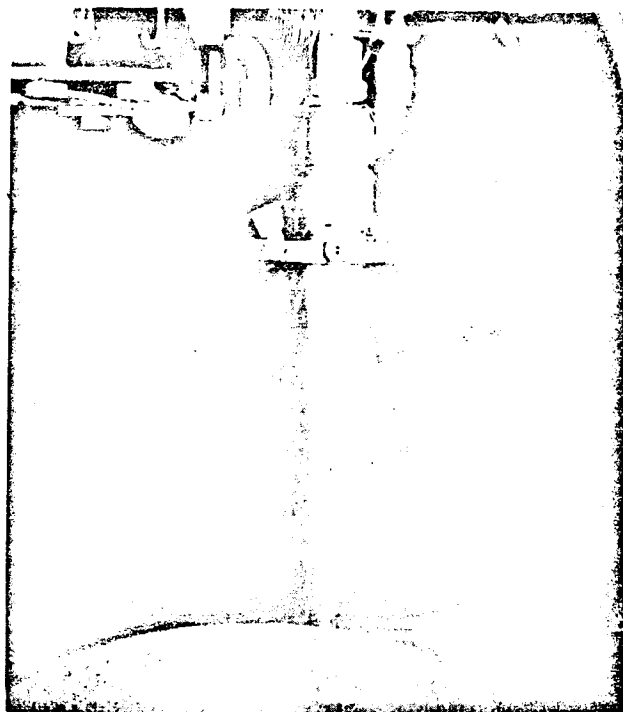
Figure 37. Injector Flow Pattern (With Splashplate)

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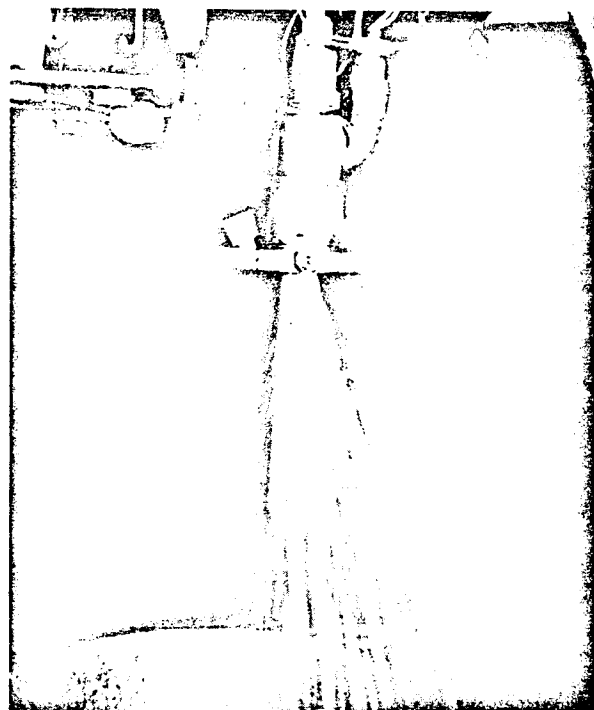
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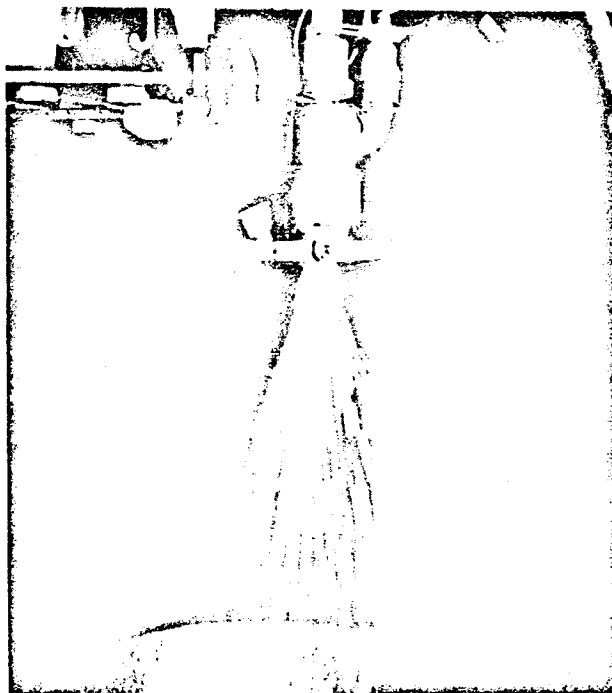
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Oxidizer only



Fuel only



Oxidizer and Fuel

Figure 38. Injector Flow Pattern (With Splashplate)  
Rotated 90 Percent

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APPENDIX H

LOW SUPPLY PRESSURE TEST

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TITLE: LOW SUPPLY PRESSURE DESIGN INFORMATION TEST

INTRODUCTION

The low inlet supply pressure design information test was completed on June 5, 1964 at SSFL, CTL III, Cell 390, with a 100 lb. flightweight thrust chamber of the Capsule II design. The test consisted of at least five tests at inlet supply pressures of 75, 50, and 25 percent of nominal. The tests were conducted under conditions of local ambient temperature and pressure and had a duration of five seconds each. The test objective was to determine the effects of the low propellant inlet supply pressure on the thrust chamber operation, including C\* performance, mixture ratio changes, and thrust chamber life.

SUMMARY

The low inlet supply pressure design information tests show that engine performance was good for the calibration and the 75 and 50 percent tests at a mixture ratio of 1.6. A sharp decrease in performance (and mixture ratio) is evident at 25 percent of the inlet pressure. The first three liners at the injector end of the chamber were lost at the completion of the 25 percent tests. This has been attributed to the looseness of these liners which was evident prior to the start of testing. X-rays revealed no damage to the ablative portion of the thrust chamber. Total accumulated burntime on the thrust chamber is 201.6 seconds.

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## DISCUSSION

### Chamber Configuration

The TCA used for the low supply pressure DIT was a 100 lb. flightweight thrust chamber P/N 207540-41, S/N 4053753 of the Capsule II design. The chamber was converted for use as test only because of two longitudinal cracks in the throat insert and a crack in the liner segment at the  $P_c$  hole. The liners at the injector end of the chamber including the cracked liner, were found to be loose during the calibration tests which preceded the low supply pressure testing.

## TEST DESCRIPTION

Fifteen low supply pressure tests were run. These tests consisted of five steady state tests of five seconds duration, at inlet pressures of 75, 50, and 25 percent of the design nominal inlet supply pressure. The design nominal inlet pressures at 100 percent are 269.2 psig for the oxidizer, and 272.2 psig for the fuel. High response transducers (Data Sensors) were used to record chamber pressure, and were calibrated prior to and immediately following each days testing or whenever suspicious data was obtained. Due to the low flowrates of propellant, 25 lb. Cox (6-0 type) flowmeters were substituted for the 100 lb. Cox (6-3 type) flowmeters for the 25 percent tests. The TCA was fired in a vertical down position under conditions of local ambient temperature and pressure.

## TEST RESULTS

The test results showed the engine performance for the calibration tests to be  $C^* = 5360$  ft/sec with an efficiency of 91.8 percent at a mixture

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ratio of 1.6. These tests were at a simulated altitude of 100,000 ft. while the low supply pressure tests were at ambient conditions. Performance for the 75 and 50 percent low supply pressure tests remained within  $\pm 2$  percent of the calibration test results but the efficiency for the 25 percent tests decayed to 85 percent (Figure 39). A mixture ratio of 1.6 was maintained for the 75 and 50 percent tests but dropped considerably (1.35) for the 25 percent tests (Figure 40). The inlet pressures were maintained at their respective percentage of design pressure within a tolerance of  $\pm 5$  psig. Figure 41 is a plot of chamber pressure versus characteristic velocity. The tests were conducted in a descending order of inlet supply pressure. The 75 percent tests were followed by the 50 percent and then the 25 percent tests. Upon completion of the 25 percent tests, the first three liners at the injector end were lost. Since only three of the 50 percent tests concluded in acceptable data, an attempt to repeat these tests was made but resulted in a marked increase in engine performance. This increase in performance is attributed to the change in burn characteristics at the head end of the chamber. Previous R&D tests on birdcage billets without liners have shown this tendency to have a higher performance but a shorter life. Audible screech was present during the 25 percent inlet supply tests. Valve response both on opening and closing was normal throughout the tests. Post test X-rays of the TCA showed the ablative body to be in excellent condition with no delaminations. The loss of the liners is attributed to their looseness in the chamber rather than as a result of the low supply pressure tests.

#### CONCLUSIONS

The low supply pressure test showed that the Capsule II type thrust chamber operated at inlet supply pressures as low as 50 percent of

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nominal will not result in any degradation of performance or life of the TCA. However, supply pressures of below 50 percent show a marked decrease in performance. No structural failures were evident as a result of these tests either in the combustion chamber or to the ablative body. The loss of the liners at the injector end at the conclusion of the 25 percent tests has been considered to be a result of their looseness rather than as a result of the low supply pressure tests.

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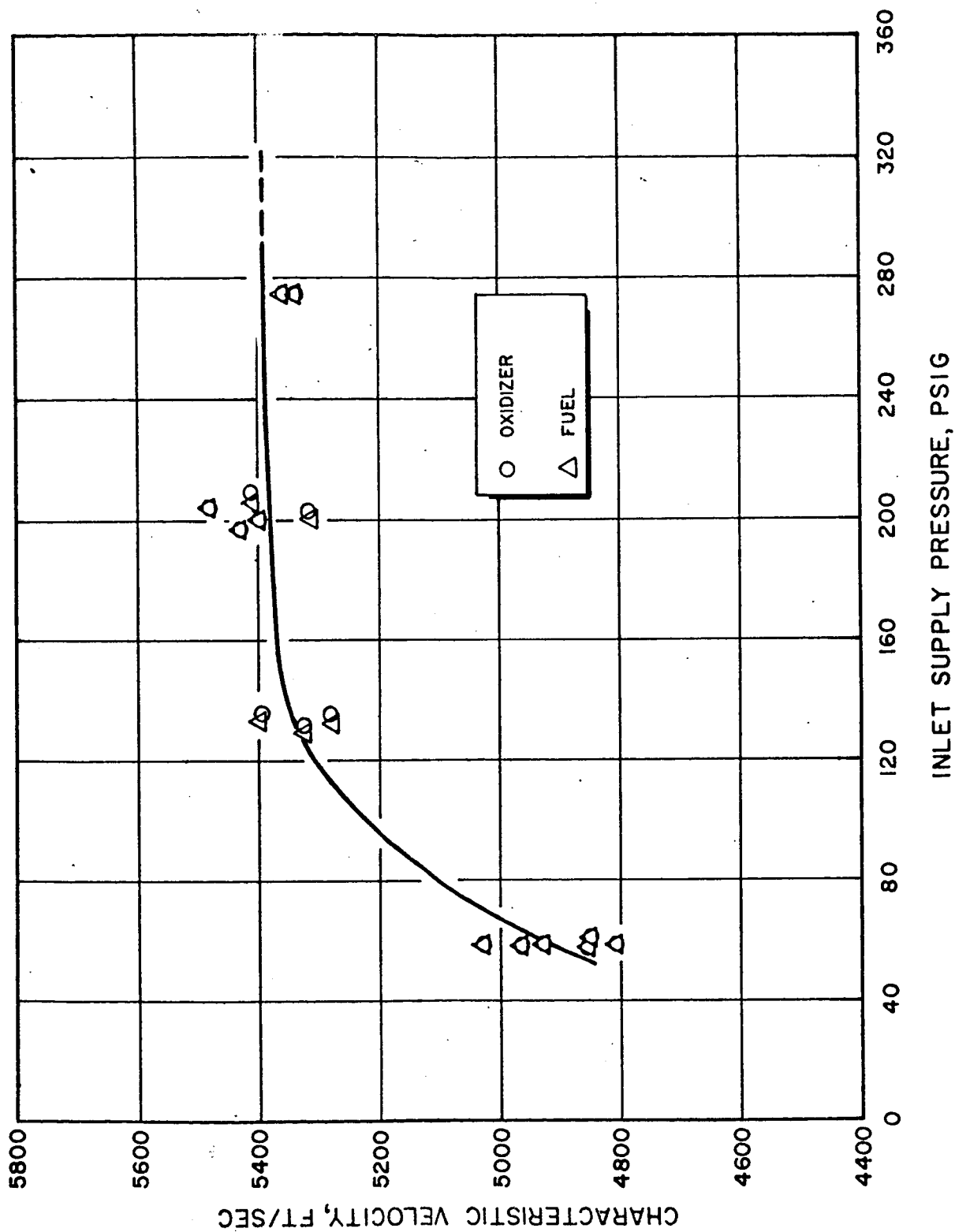


Figure 39. Characteristic Velocity vs Inlet Supply Pressure, Low Supply Pressure DIT

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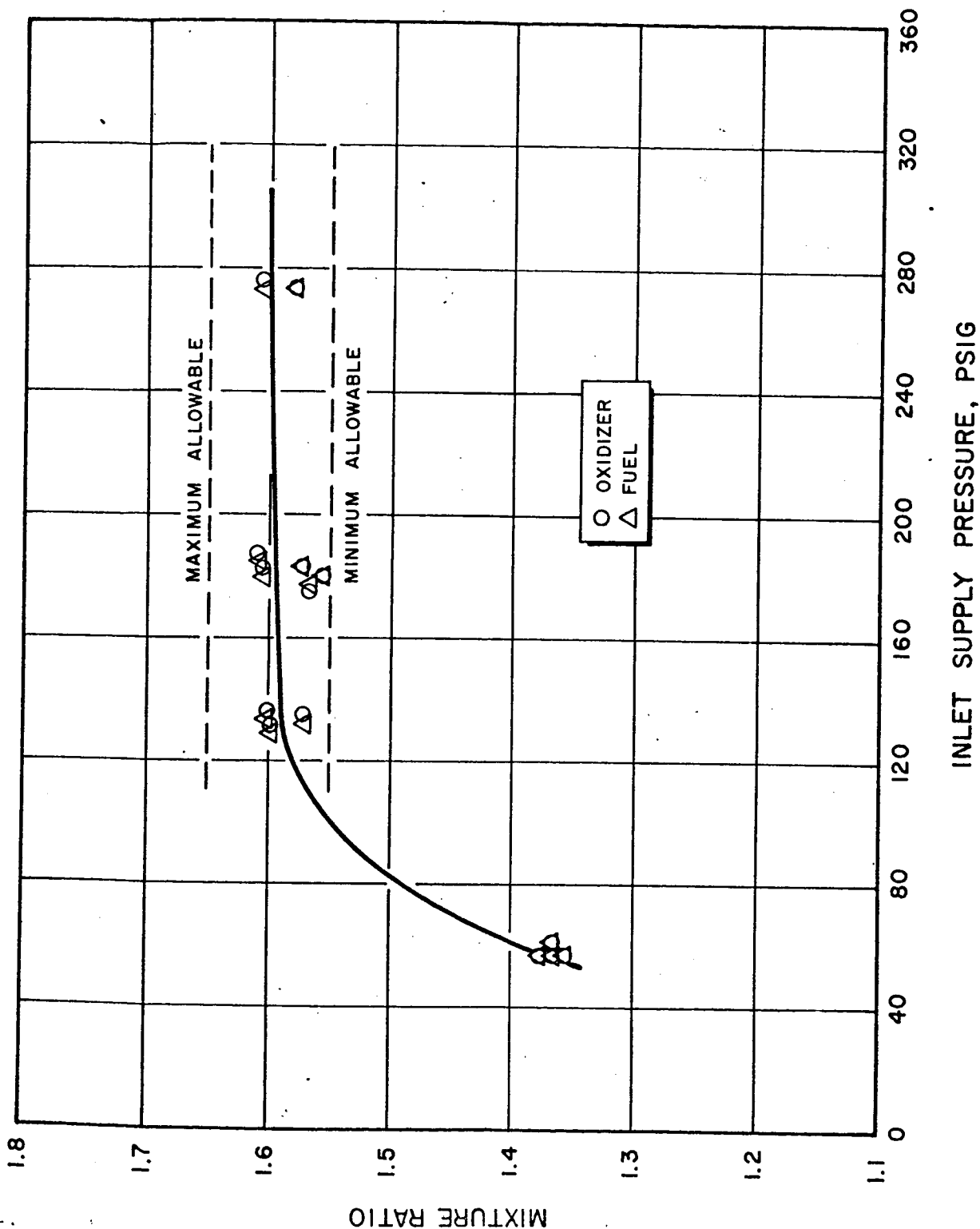


Figure 40. Mixture Ratio vs Inlet Supply Pressure, Low Supply Pressure DIT

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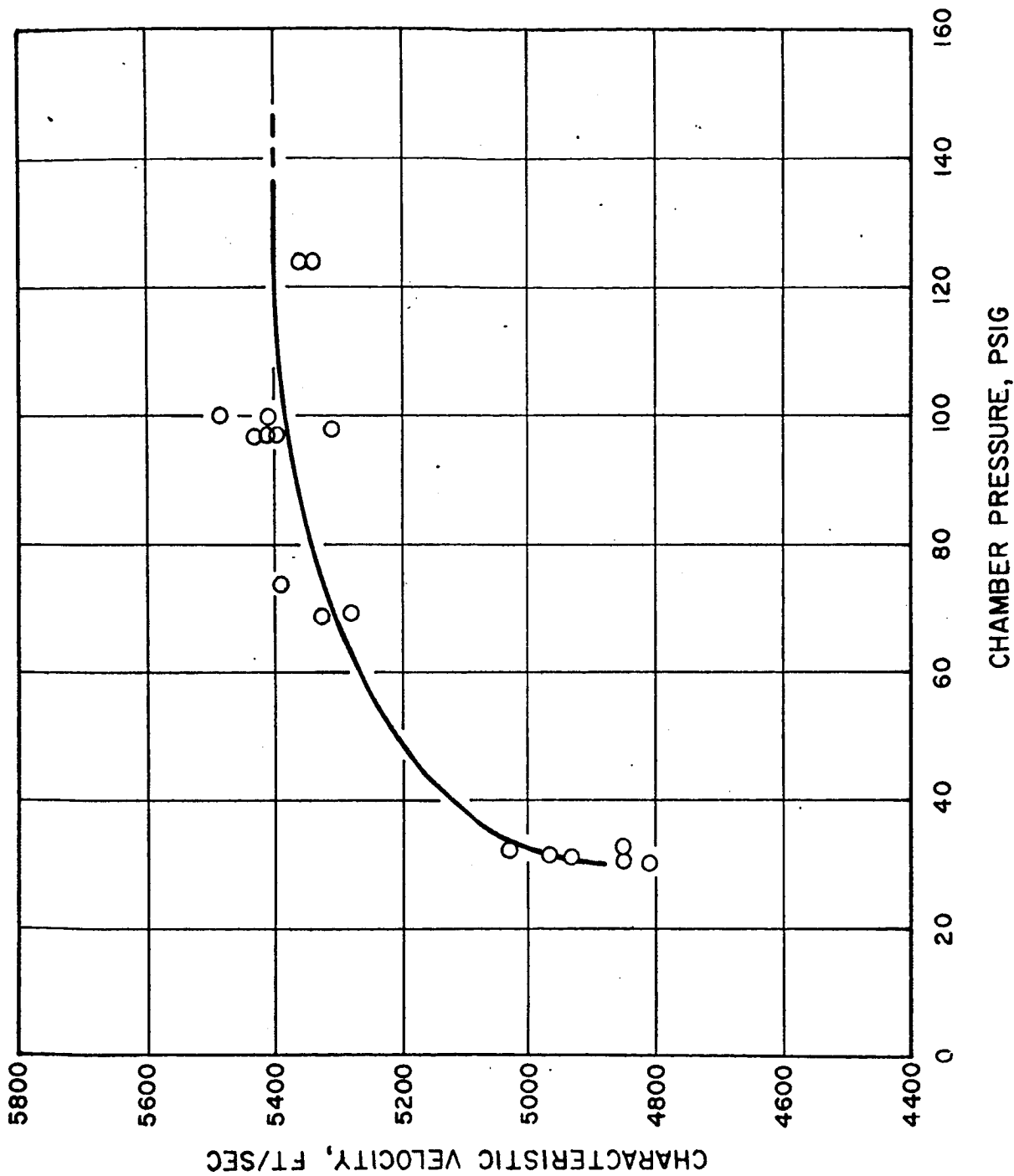


Figure 41. Characteristic Velocity vs Chamber Pressure,  
Low Supply Pressure DIT

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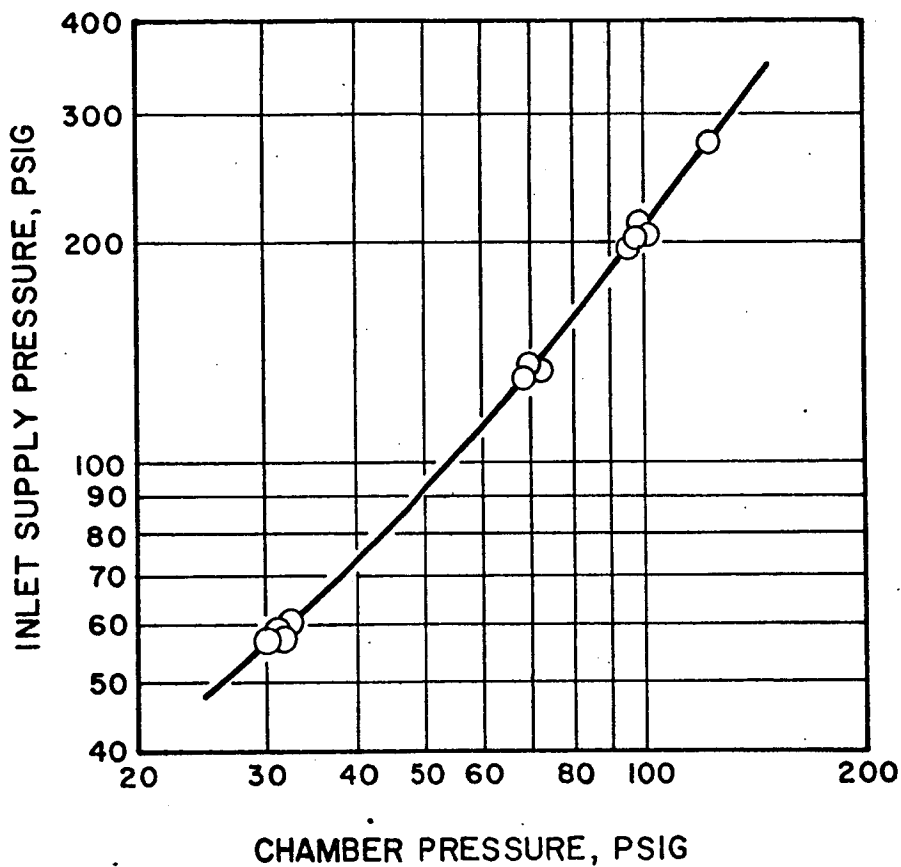


Figure 42. Inlet Supply Pressure vs Chamber Pressure,  
Low Supply Pressure DIT

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